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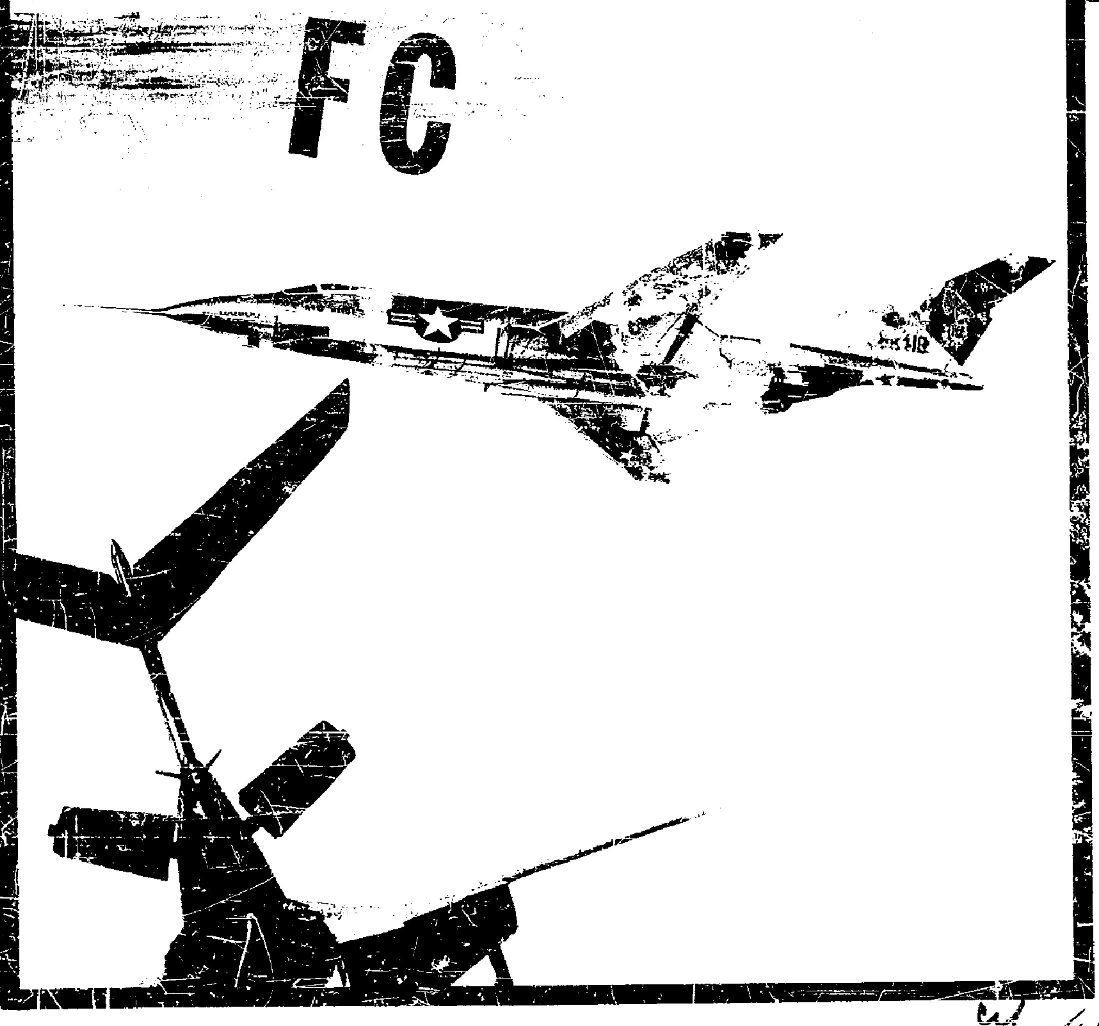
F-101A

AFFTC-TR-5' 32
SEPTEMBER 55

JACK STRIER
PROJECT ENGINEER

USTIN A. JULIAN, JR. MAJOR, USAF
PROJECT PILOT



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AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR RESEARCH AND DEVELOPMENT COMMAND

55AA 45695

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OCT 20 1955

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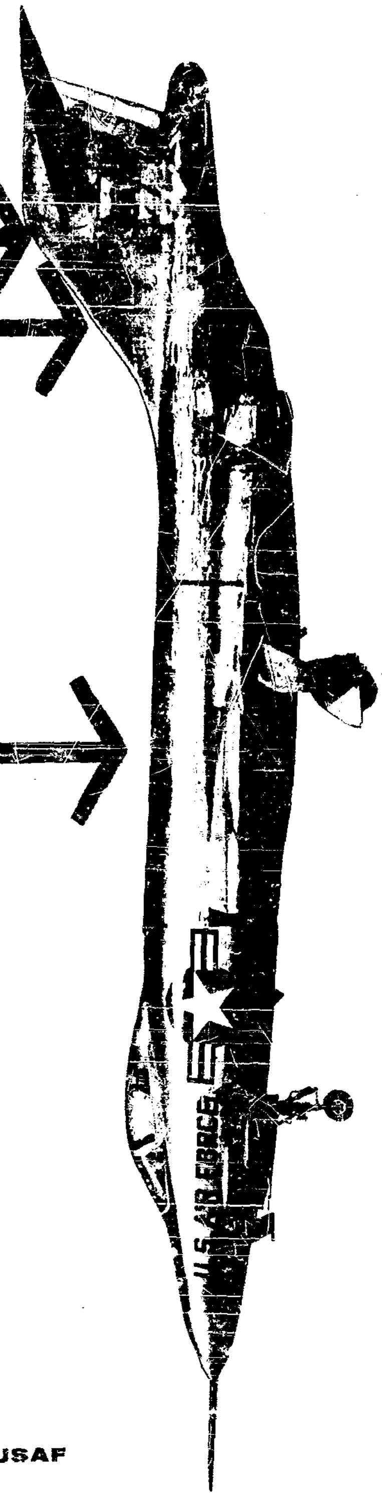

F-101A PHASE II FLIGHT EVALUATION

AFFTC-TR-55-32
SEPTEMBER 1955

JACK STRIER
PROJECT ENGINEER

AUSTIN A. JULIAN, MAJ. USAF
PROJECT PILOT

55AA 45695

CONFIDENTIAL

this report has been reviewed and approved

H. A. Hanes J. S. Holtoner

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Director, Flight Test

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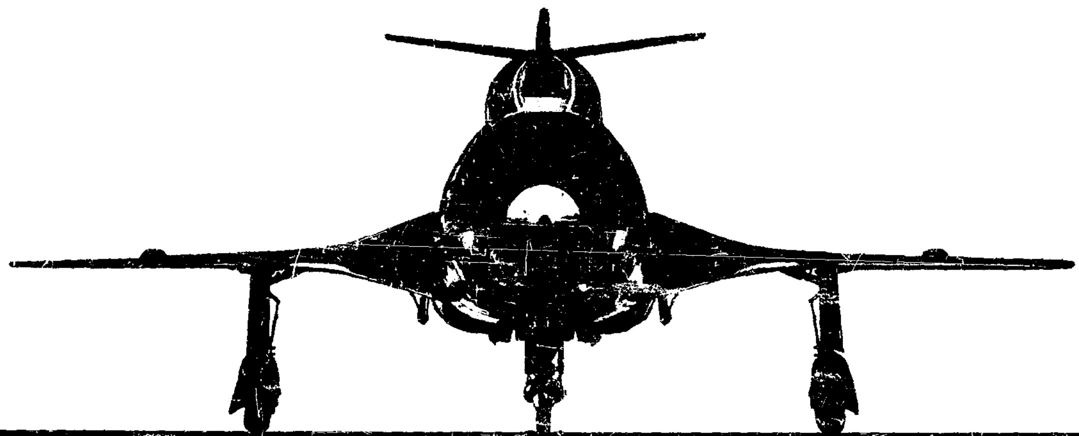
Commander

abstract

The F-101A is potentially a good multi-purpose fighter. It is unacceptable for service use at the present time because of comparatively poor high altitude performance, restricted maneuverability due to engine compressor stalls under accelerated flight maneuvers, and lateral control sensitivity. The rate of climb, level flight speed and ability to accelerate in level flight are good but the 49,200-foot service ceiling is poor in comparison with USAF fighter aircraft in operational use. The range capabilities with the present bleed valve schedule are 15% less than predicted values.



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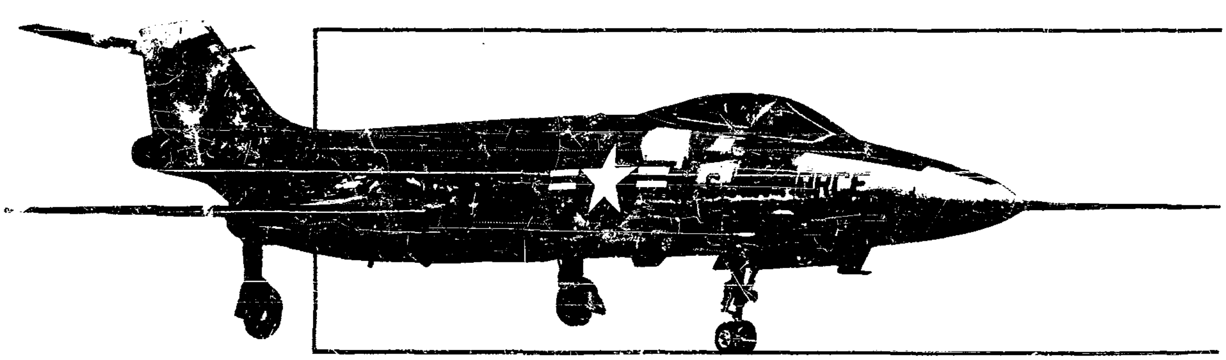
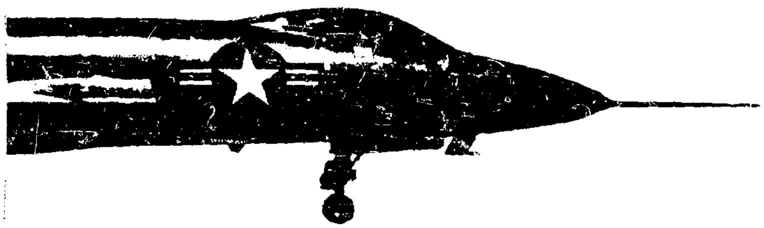
** Due to the limited requirement for the information presented in this Appendix, it has been published as a separate volume. Copies of this Appendix can be obtained by writing to the Armed Services Technical Information Agency (see inside front cover).*



INTRODUCTION

This report presents
the results of the Phase II
Flight Test conducted
on the F-101A aircraft,
USAF No 53-2419.
The purpose of this test
was to obtain data
on the performance,
stability and control
of the aircraft,
and to secure pilot's comments
regarding handling characteristics.
The test was conducted
under the authority of
Air Research and Development
Command Test Directive No. 5389-F1
dated 15 June 1954.
The flight test program,
consisting of 32 flights
totaling 30:20 hours,
was conducted at the
Air Force Flight Test Center,
Edwards Air Force Base, California,
between 28 April
and 26 May 1955.

The F-101A is a swept wing, single-place fighter powered by two Pratt and Whitney engines equipped with short afterburners. These engines are currently designated YJ57-P-13 and are rated at 14,000 pounds thrust with afterburners in operation. The internal fuel capacity of the aircraft is 2,000 gallons. Armament includes four fixed forward-firing M-39 20mm guns and provision for external fuselage-mounted stores. Stabilator, mid-span ailerons, and rudder are controlled by conventional stick and pedals, and powered by two independent, irreversible hydraulic control systems with artificial feel systems.



Modifications which were made to the test aircraft prior to Phase II tests as a result of Air Force preliminary evaluation flights during Phase I, or as a result of the contractor's experience on the initial airplane are as follows:

A revised inlet duct configuration was installed in aircraft number 12 and subsequent aircraft to improve stall characteristics.

Pratt and Whitney, "Gold Plated," YJ57-P-13G modified engines with an increased stall margin were installed.

The heat and vent system was modified to provide improved flow stability and to eliminate the objectionable amount of air blowing into the pilot's face.

Canopy-windshield latches were incorporated to eliminate the deflection of the canopy relative to the windshield at higher speeds.

The longitudinal maneuvering stick forces were reduced appreciably, particularly in the supersonic Mach number range.

Objectionable speed brake buffet was reduced by decreasing the full deflection brake extension to 50 degrees.

The wheel brake pedal deflection was reduced for full brake application.

The yaw damper was made operative to improve the lateral-directional dynamic stability and to reduce adverse yaw during roll maneuvers.

The intensity of cockpit warning lights was increased appreciably.

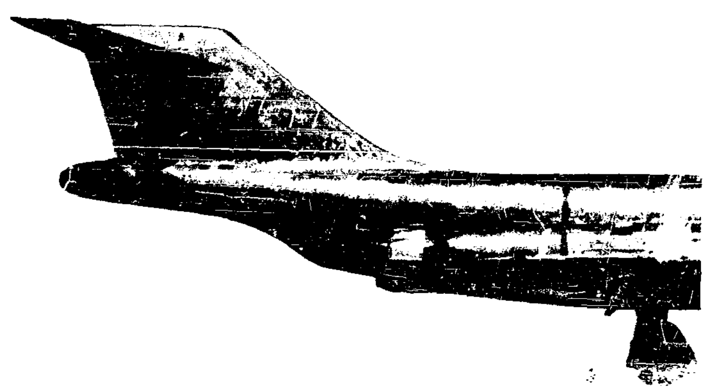
Cockpit lights were installed to indicate take-off trim position of the longitudinal and directional control surfaces.

A lateral stick bob-weight was installed to eliminate the lateral control system instability which developed following stick "raps" at approximately 0.85 Mach number.

The aft fuselage "heat blanket" was changed to eliminate buckling and fatigue deterioration and to provide improved protection of the structure.

Engine oil fumes in the cockpit were eliminated by changing the location of the bleed for cabin air supply. Specific evaluation of the requested improvements are made in the applicable section of this report.

Performance of the aircraft was evaluated for an engine start gross weight of 40,530 pounds. This includes a full fuel load of 2,000 gallons. Stability, control and handling characteristics were evaluated for the normal take-off center-of-gravity of 31.4% MAC. The center-of-gravity moved aft approximately 5.6% MAC from the take-off position as internal fuel was consumed. Weight and balance for all flights are presented in Appendix II. Phase II flight limitations placed on the F-101A airplane by the contractor are listed on page 203, Appendix II. These limitations were placed on the aircraft until all flight operating boundaries of the aircraft have been investigated. This prevented investigation of inertia coupling characteristics during the Phase II test.

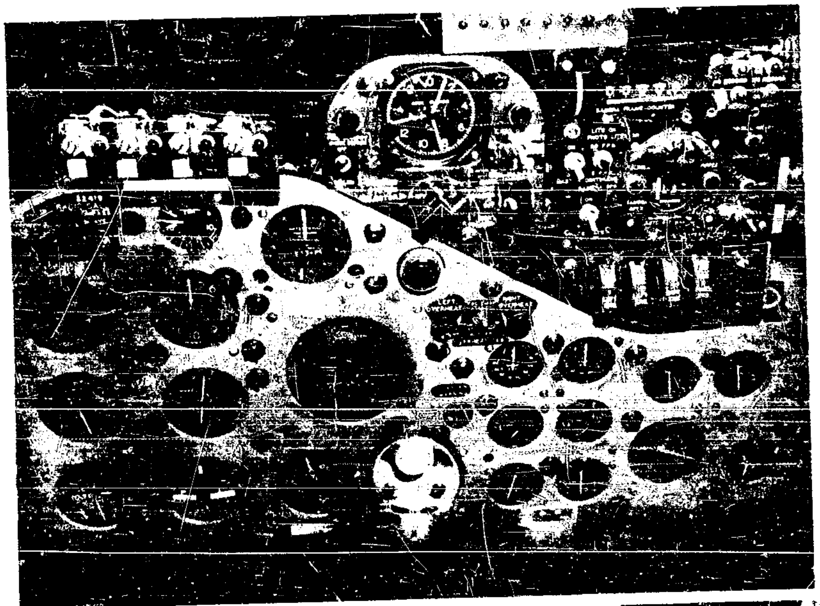


TEST RESULTS performance

■ cockpit evaluation

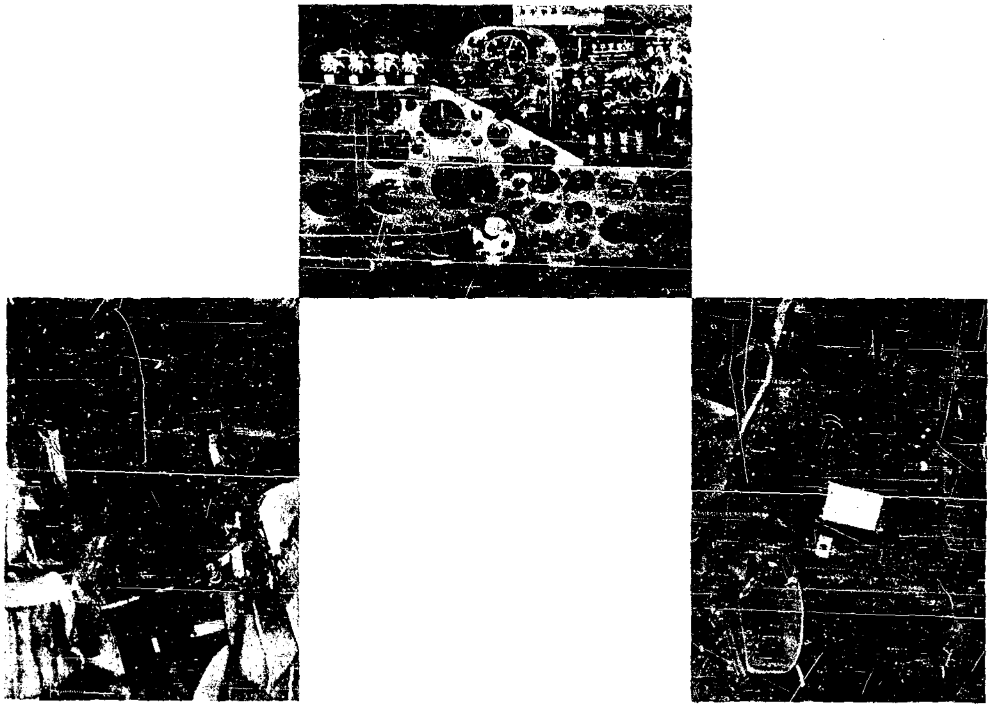
Entrance to the cockpit is gained by use of the ladder that hooks over the edge of the cockpit. No kick steps or hand grips are provided and are not considered necessary due to the height of the aircraft from the ground to the bottom line of the fuselage. Visibility from the cockpit is excellent except that it is difficult to see the aileron positions when trimming for take-off. The intensity of all warning lights in the cockpit and the intensity control are adequate. The seat and rudder adjustments and seat back angle are satisfactory. Generally, the cockpit arrangement is considered satisfactory.

Left side console



Center console

Right side console



Items of cockpit arrangement which are considered unsatisfactory are as follows:

1 The side consoles are located below the pilot's normal reach when his seat is adjusted for proper height. The consoles are so low that lighting is inadequate in daylight making it very difficult for the pilot to identify various instruments and instrument controls without bending his head down, inside the cockpit, to make positive identification. The consoles should be raised approximately three inches and tilted inward an additional five degrees. This location would be more perpendicular to the pilot's line of vision and would enable him to identify the console instruments and to operate the console controls easily.

2 The throttle quadrant is still completely unsatisfactory. This item was discussed in the preliminary evaluation of the aircraft. The only modification accomplished prior to Phase II was closing the distance between the finger lifts. The throttles are much too large and cumbersome for the average pilot. The use of the finger lifts to control both afterburner and throttle motion requires an overly complex manual operation that could lead to results other than those desired. It is impossible to operate the throttles, finger lifts, take-off locks and nose wheel steering simultaneously.

3 The microphone antennae selector switch, located on the right throttle, seems an unnecessary complication to radio operation. The position of this switch selects the antenna to be used, upper or lower, and thus requires the pilot to know the relative position of the desired target of his transmission. The switch is spring loaded to the center (off) position and is difficult to operate. A simple push-type microphone button located on the throttle would be much more desirable. If two antennae are required, a separate antennae selector switch located elsewhere would be desirable.

4 The spring loaded speed brake switch is located too low on the side of the throttle. Operation of the switch requires an unnatural thumb motion and the switch must be held in the open position until the brake travel has been completed. This is not desirable.

5 The speed brake "open" warning light is difficult to see because of its location on the center portion of the left side console just aft of the pilot. It is quite possible for the pilot, when using speed brakes during formation flying or "join-up," to inadvertently leave the speed brakes partially cracked without noticing the light. There is a small amount of airframe buffet in certain subsonic speed ranges. The pilot could very easily attribute the speed brake buffet to normal airframe buffet.

6 The speed brake switch and microphone switches are identical and are located very close to each other. On numerous occasions, the pilot actuated the speed brake switch instead of the microphone switch.

7 There is no minimum afterburner detent or latch to prevent the pilot from inadvertently coming out of afterburning when modulating power in formation.

8 Rudder and stabilator trim for take-off are indicated by individual lights. They are located on the vertical panel of the left console under the canopy rail and are difficult to see. This location is not desirable but is acceptable. The rudder trim light will illuminate more than a three-degree spread of the neutral rudder position, therefore when the light is on it does not necessarily mean that the rudder is trimmed for take-off. There is no aileron trim light installed. Aileron trim for take-off has to be done visually. Once the pilot is strapped into the cockpit it is very difficult to see the ailerons. A more positive aileron trim indication is needed since the aircraft is very sensitive laterally after take-off. Pilot convenience and ease of operation would be improved if a single light incorporating stabilator and rudder trim as well as aileron trim were installed. When appropriate trim is being actuated this single light would come on as each control surface passes through the take-off trim position. If practical, a single control button, which would place all trim in take-off position when depressed, would be satisfactory. The desired system is presently installed in the F-100 type aircraft.

9 Hydraulic power is supplied by two completely separate systems, the primary power control and utility system. This consists of four hydraulic pumps, one primary and one utility pump operated from the left engine; and one primary and one utility pump operated from the right engine. Cockpit hydraulic pressure indication consists of one utility pressure gage and one primary pressure gage. With the present installation and both engines operating it is possible to lose one utility pump and one primary pump without the pilot being aware of it. It is felt that the failure of a hydraulic pump is a critical item. A means should be provided to give the pilot a positive indication that all hydraulic pumps and systems are operating properly.

10 The landing gear warning buzzer is too loud. The noise level of the buzzer could be cut down one-third and still serve its intended purpose.

11 The emergency brake control is located on the lower right portion of the panel. In the event conditions should arise necessitating its use, the pilot would be required to release the control stick with his right hand in order to actuate the emergency brake control As production models will incorporate nose wheel steering on the control stick, the nose wheel steering, normally operated by the right hand, would have to be actuated by the left hand during emergency brake operation. To avoid switching hands and to eliminate an accident hazard, the emergency brake control should be relocated to the left side of the panel.

12 The throttle friction level located between the throttles is awkward and difficult to operate.

13 The nose wheel steering control is located on the left throttle. This is a very undesirable location and the control is very awkward to operate.

14 There is no parachute support behind the pilot for a back parachute; therefore the weight of the parachute must be carried by the pilot during the entire flight.

15 The needle and ball instrument is frequently used in retrimming the aircraft, but it is off center to the left from the pilot's normal line of sight. A quick reading gives an erroneous impression of trim conditions.

taxiing and ground handling

The F-101A is difficult to taxi, primarily because of very high rudder forces. An estimated force of 200 to 300 pounds is required for full rudder control deflection and to obtain the specified forty degrees of nose wheel steering. It is felt the forty degrees of nose wheel steering is not being attained in the F-101. Nose wheel steering requires too much rudder control deflection to obtain the desired turning radius.

The poor nose gear centering characteristics also contribute to the taxiing difficulty experienced with the F-101A. When the nose wheel steering system is engaged by the pilot, with the rudder controls in a neutral position, the aircraft tends to pull to the left, requiring constant pumping of the right rudder in order to taxi in a straight line. The easiest way to counteract the pull to the left is to taxi with the left engine.

The combination of high rudder forces, turning radius and poor nose gear centering encourages the pilot to disregard nose wheel steering and use wheel brakes. Excessive use of the brakes promotes increased brake and tire wear. An average of only there or four landings per tire was obtained during the Phase II flight tests, however the majority of flights, were conducted from the dry lake bed, which required excessive taxiing.

The nose wheel steering button is located on the left throttle quadrant. Some difficulty is experienced in holding the nose wheel steering button to "engage" when manipulating the throttle.

While taxiing a very loud clanking is heard each time the nose wheel passes over the slightest rough spot. The noise is due to the vertical or side play in the trunnion mount bearing of the nose gear drag brace. The clanking can be eliminated by installing a shim which reduces the play.

Wheel braking for taxi operation is adequate. The angle of brake pedal rotation on maximum deflection has been improved to where it is now satisfactory. Although braking forces are greater than those required on the F-100 type aircraft the brakes are considered satisfactory.

The canopy is electrically operated with the canopy switch located on the left side of the cockpit just under the canopy rail. No mechanical locks are provided. The system is so arranged that when the nose gear is turned to either side or retracted, the canopy switch is deactivated. The reason for this is to prevent inadvertent actuation of the canopy switch during flight. This reason is not considered valid and the situation is very unsatisfactory. The pilot has no means of getting rid of the canopy during flight or a wheels-up landing other than by blowing it off. If this fails, then the pilot is forced to eject through the canopy. The pilot could be stunned in the course of a wheels-up landing, and may, in the attempt to eject the canopy, inadvertently eject himself.

take-off and initial climb

All take-offs were made with wing flaps fully extended. There is no intermediate or hold position. Take-offs were accomplished using both military and maximum power. The majority of take-offs, with the exception of afterburner climbs and maximum performance take-offs, were made with military power to insure a reasonably closed afterburner nozzle at altitude.

The wheel brakes can be locked when maximum power is being used; however, with the large amount of thrust available the aircraft will skip along the runway, resulting in unnecessary damage to the tires. The technique used for maximum power take-offs was to lock the brakes until full military power was set, then the brakes were released and the afterburners lit individually while the aircraft accelerated. Although rudder control becomes effective at approximately 60 knots IAS, aircraft response is slow, and directional control should be maintained by use of nose wheel steering up to approximately 95 to 100 knots IAS. The nose wheel lift-off speeds were between 135 and 140 knots IAS, the aircraft becoming airborne at 150 to 155 knots IAS. The nose gear can be rotated at 110 to 115 knots IAS with the use of full stabilator; however, the aircraft acceleration to its take-off speed of 150 to 155 knots IAS is slower than if the aircraft is allowed to accelerate to 135 to 140 knots before the

nose wheel is lifted off. After take-off, acceleration is rapid and a relatively steep climb-angle must be assumed. Gear and flap retraction must be initiated immediately in order to prevent exceeding the maximum gear-down limit speed of 250 knots and the maximum flap-down speed of 290 knots. The slight nose up trim change encountered presents no problem. The forward visibility is good during the transition time from unstick to gear and flap retraction. Longitudinal control is satisfactory and stick forces are light. Directional control is good but the rudder forces are too high. The lateral control is extremely sensitive resulting in a tendency to overcontrol. The problem is more acute when turbulent air and gusty winds are encountered. Lateral control during take-off and initial climb is unsatisfactory. The aircraft acceleration to best climb speed is good.

Two photo-recorded maximum power take-offs were made from a concrete runway. This data corrected to sea level, no wind, standard NACA atmospheric conditions, is presented in the following table:

Power	Gr. Wt. — lb	Ground Roll — ft	Total Dist. to 50' ft	Nose Boom Indicated Airspeed NWO	knots T.O.	50'	Std. System Indicated Airspeed NWO	knots T.O.	50'
Max.	40,890	2335	4010	138	150	184	143	152	188.5
Max.	40,660	2575	4655	104	148.5	184	111	150	189

Cross wind take-offs were accomplished in gusty winds up to 35 knots. Cross wind take-offs are very difficult and control is unsatisfactory due to the poor nose wheel steering, high rudder forces and lateral control sensitivity after becoming airborne. Normal procedure to correct for cross winds during the ground roll is to use nose wheel steering. In the absence of adequate nose wheel steering, directional control can be maintained by brakes; however, brake response is quite sensitive and the pilot is prone to overcontrol. Cross wind take-off characteristics should be satisfactory when the nose wheel steering is improved, rudder forces reduced and lateral control sensitivity decreased.

Take-off auxiliary position was utilized on two occasions. Use of take-off auxiliary locks requires pushing down on the individual finger lifts at the military power position and then pushing the throttles forward approximately two inches before firing the afterburners. This places the throttles very close to the landing gear handle resulting in a very difficult landing gear handle operation. With the throttles in this position it is almost impossible to manipulate the finger lifts and nose wheel steering with the shoulder harness locked. Even with throttles at the military power setting this same difficulty is experienced; therefore use of the auxiliary position was discontinued. For normal operation it is felt that the single engine capabilities of the aircraft are such that take-off locks are not a necessity. This is further justified as in the case of a formation take-off. If the locks were used neither aircraft would be able to modulate power as is necessary in a formation take-off. This should not be interpreted to mean that the take-off locks are not required for heavy gross weight conditions with external tanks or stores.

climbs

Continuous climbs were flown to the cruise ceiling of the aircraft at maximum and military power. No sawtooth climb tests were accomplished, and climb speed schedules used were those provided by the contractor.

Maximum Power (afterburning): The acceleration of the F-101A after take-off to best climb speed is rapid, taking approximately 1.13 minutes from brake release to best climb speed. After the desired climb speed is reached the nose is pulled sharply up to maintain the prescribed climb schedule. Up to 40,000 feet, the climb angle is such that the horizon cannot be solely utilized as a reference, necessitating partial use of the artificial horizon as a cross reference. The climb speed schedule is easiest to maintain by flying with the aid of the Mach number indicator. The aircraft is very sensitive laterally and it is difficult to keep from overcontrolling. Two of the three maximum power climbs flown were discontinued when the right afterburner blew out and subsequent compressor stalls were experienced. The other climb was terminated by right engine compressor stalls. When the climbs were terminated, the aircraft was near its cruise ceiling. Maximum power climb performance up to 45,000 feet is excellent in comparison with other fighter aircraft; however, the rate of climb drops off rapidly with altitude and a relatively low service ceiling of 49,200 feet is reached after 7.1 minutes from sea level.

Military Power Climb: Using military power, the rate of climb is much lower than the maximum power climbs; consequently the angle of climb is not as steep and visibility is improved. The climb speed schedule is easily maintained with either the use of the Mach number or airspeed indicator. The lateral control sensitivity problem is increased slightly with the lower climb speeds. The military power service ceiling is 44,000 feet. The best climb speeds at altitudes close to the cruise ceiling are on the edge of the buffet region.

Single Engine Maximum Power Climb: A single engine maximum power climb was flown by leaving one engine in idle power. The resulting rate of climb is higher than if the engine control was in idle cut-off and the engine windmilling. The climb was discontinued before the cruise ceiling was reached because of cloud coverage and reduced visibility.

A summary of the climb performance of the F-101A is presented in the following tables. This data, corrected to standard NACA atmospheric conditions, is presented in Figures 1, 2 and 3, Appendix I.

MAXIMUM POWER CLIMB PERFORMANCE

Gross Weight at Engine Start — 40,530 Pounds

Altitude ft	R/C ft/min	T/C** min	Fuel* Used lb	Gross Weight lb	Distance Traveled naut. mi.	True A/S knots
S.L. (est)	29,000	0	0	39,000	0	560
10,000	25,300	.37	419	38,581	3.5	580
20,000	20,650	.80	826	38,174	7.6	570
30,000	15,000	1.36	1236	37,764	12.9	548
40,000	7,550	2.27	1710	37,290	21.1	540
45,000	3,250	3.25	2100	36,900	29.5	540
49,200 S/C	100	7.10	3460	35,540	64.0	545

* For maximum power climbs allow 1530 pounds of fuel for start, taxi, take-off and acceleration to best climb speed.

** To obtain time to climb from brake release add 1.13 minutes.

MILITARY POWER CLIMB PERFORMANCE

Gross Weight at Engine Start — 40,530 Pounds

Altitude feet	R/C ft/min	T/C** min	Fuel* Used lb	Gross Weight lb	Dist. Traveled naut. mi.	True A/S knots
S.L. (est)	11,000	0	0	39,460	0	...
10,000	10,200	.94	260	39,200	7.7	504
20,000	8,875	1.98	501	38,959	16.4	506
30,000	6,500	3.26	734	38,726	27.1	493
40,000	2,525	5.56	1046	38,414	45.8	490
44,000 S/C	100	9.70	1460	38,000	...	490

* For military power climbs allow 1070 pounds of fuel for start, taxi, take-off and acceleration to best climb speed.

** To obtain time to climb from brake release add two minutes.

SINGLE ENGINE MAXIMUM POWER CLIMB PERFORMANCE

(Right Engine Operative; Left Engine in Idle Power)
Gross Weight at Engine Start — at 40,530 lb

Altitude feet	R/C ft/min	T/C min	Fuel* Used lb	Gross Weight lb	Distance Traveled naut. mi.	True A/S knots
S.L. (est)	7,200	0	0	39,460	0	...
10,000	6,800	1.42	726	38,734	9.5	409
20,000	5,950	2.98	1372	38,088	20.2	407
30,000	3,070	5.00	2004	37,456	33.6	393

* For single engine maximum power climbs allow 1070 pounds of fuel for start, taxi, take-off and acceleration to best climb speed.

The climb performance of the F-101A is briefly compared in the following table with similar data for several currently operational and prototype aircraft. All data is presented for maximum power and with no external stores except that of the F-94C, which carried tip tanks, and the F-86H, which used military power.

	Gross Weight at S.L. lb	R/C at S.L. ft/min	R/C at 10,000 ft	R/C at 30,000 ft	Combat Ceiling 500 ft/min	T/C to* 45,000 ft
F-101A	39,000	29,000	25,300	15,000	48,600	3.25
XF-104 AF Tech Rpt 55-8	15,610	32,000	26,100	13,850	48,650	3.15
F-100A AF Tech Rpt 54-26	24,000	20,000	19,600	13,500	54,200	3.40
F-94C AF Tech Rpt 53-30	19,500	9,500	7,800	4,500	47,000	10.5
YF-102A AF Tech Rpt 55-31	26,300	17,300	16,900	13,075	53,500	3.65
F-86D** Addendum to Rpt 53-26	16,800	12,800	11,900	7,000	49,000	6.7
F-86H AF Tech Rpt 54-4	19,000	12,000	9,700	5,300	47,500	9.00

* This does not include the time required to accelerate to best climb speed from brake release.

** Rate of climb is for the flat top fuel schedule B-4098004-706.

level flight

Maximum Speeds: Maximum level flight speeds of the F-101A within its operational altitude are exceeded only by the XF-104. The level flight acceleration characteristics of the F-101A are superior to those of the XF-104. For all practical purposes the F-101A at 35,000 feet can accelerate from .85 Mach number to its maximum level flight speed in approximately 5 minutes. Acceleration is good at all altitudes. Supersonic flight was attained at altitudes above 13,000 feet. Supersonic formation flight will require varying throttle setting while in afterburning. This will require extreme care on the part of the pilot as there is no minimum afterburning stop or detent to prevent his inadvertently coming out of afterburning. On the preliminary Phase I evaluation flights the stick position was too far aft for supersonic flight. The stick position has been modified to where it is satisfactory. For supersonic flight, the longitudinal trim rate is satisfactory and the aircraft is very easy to trim out. Maximum level flight speeds were determined over a range of altitudes. This data, corrected to a standard NACA atmosphere and a gross weight of 35,000 pounds, is presented graphically in Figures 4 and 5, Appendix 1, and is summarized in the following table:

	F-101A Phase II Results				McDonnell Report No. 3683			
	Military Power		Maximum Power		Military Power		Maximum Power	
Alt. ft	Mach No	True A/S kts	Mach No	True A/S kts	Mach No	True A/S kts	Mach No	True A/S kts
13,000	.934	589	1.011	638	.934	589	1.070	675
22,000	.950	578.5	1.185	721.5	.950	578.5	1.273	775.5
30,000	.963	567	1.354	797	.963	567	1.41	830
35,000	.960	552.5	1.442	830	.961	553	1.458	839
40,000	.945	543.5	1.422	818	.942	541.5	1.409	810
45,000	.885	509	1.272	731.5	.880	506	1.222	702.5

Cruise Control Data: Cruise control data corrected to standard NACA atmosphere is presented in Figures 6 through 11. The results from the limited amount of data obtained indicate that the best range of the aircraft can be realized by flying at a weight-pressure ratio of 180,000 pounds (38,530 pounds gross weight and 36,800 feet). This condition can be realized by making a military power take-off and climb up to an altitude of 36,800 feet (allowing 1070 pounds of fuel for warm-up, taxi, take-off and acceleration to best climb speed) and establishing the power for recommended cruise Mach number. This Mach number should then be maintained with constant power, gaining altitude as weight decreases. There is some indication that because of the increase in specific fuel consumption of the J57 engine at altitude, there may be an altitude-engine-airframe combination at which a longer range could be attained by maintaining a constant altitude cruise. The present intercompressor bleed valve schedule is such that the bleed valves are open in the speed range for best cruise. To determine the effect that this present schedule has upon range, cruise data was obtained at 35,000 and 40,000 feet with the bleed valves in the automatic and manually controlled positions. The manually-closed bleed valve cruise data indicated that a 6.5% increase in specific range could be realized over the range obtained with the present automatic bleed valve schedule. The specific range with the automatic bleed valve schedule was 15% less than the contractor's predicted values. Cruise control data is summarized in the following table:

RECOMMENDED CRUISE PERFORMANCE

Alt. ft	Gr. Wt. lb	Mach No	CAS kts	TAS kts	N_2 RPM	Naut. Air Miles per lb of fuel
13,000	37,000	.710	376	448	8245	.0742
30,000	38,000	.830	316	448.5	8145	.113
35,000	37,000	.815	278	469	8090	.1208
35,000**	37,000	.802	273	461.5	8010	.1272
40,000	36,500	.810	245	465.5	8275	.1255
40,000**	36,500	.815	247	468.5	8205	.134
40,000**	30,925	.805	244	463	8035	.1495
45,000	34,000	.860	234	494.5	9050	.0958

* Recommended cruising speed is the highest airspeed for 99 percent of the maximum nautical air miles per pound of fuel.

** Bleed valves closed.

Drag Data: Level flight drag data is not presented because of poor afterburner nozzle operation and the consequent unreliability of thrust measurements. Thrust data was obtained during all tests by means of a calibrated thrust rake at the standard location between the third stage turbine and the afterburner spray bar. Afterburner nozzle operation was unsatisfactory. When coming out of afterburning the nozzles would stick open at varying degrees. To close the nozzles it was necessary to reduce the power to 80% rpm and decrease the airspeed. Once the nozzles were closed they would drift open during flight as evidenced by nozzle position instrumentation and more significantly by lower airspeeds and lower P_{t7} values for a given power setting as compared to airspeeds and P_{t7} values obtained with the nozzles fully closed. A plot of $N_2/\sqrt{\theta_{t2}}$ vs. P_{t7}/P_{t2} is presented in Figure 12, Appendix I. The plot indicates the affect of afterburner nozzle position on the P_{t7}/P_{t2} pressure ratio and thrust measurement.

Engine Operation: The operation of the YJ57-P-13G engine during Phase II tests was considered unsatisfactory because of engine compressor stalls under accelerated flight maneuvers, afterburner nozzes either sticking open after coming out of afterburning or drifting open in flight, erratic bleed valve operation, and inability to relight the afterburner at altitudes of 45,000 feet and above. Though engine compressor stalls are not the problem they were with the unmodified YJ57-P-13 engines and the Nos. 1 to 11 duct configuration, they are still enough of a problem to seriously limit the operational and tactical capabilities of the aircraft. Engine compressor stalls can be encountered under 3 to 4 g positive loads regardless of speed or altitude. At altitudes above 30,000 feet, engine compressor stalls were present under accelerated flight conditions at low indicated airspeeds. Engine accelerations in level flight were considered satisfactory. Only on one occasion were compressor stalls encountered during a throttle burst. This occurred at an altitude of 40,000 feet and .825 Mach number. The afterburner would normally light at altitudes up to 42,000 feet. Lights could be sometimes obtained between 42,000 and 44,000 feet, but the afterburner could not be lit at all at 45,000 feet.

Miscellaneous: There is a very noticeable and aggravating buffet at approximately .93 Mach number. The source of this buffet should be located and eliminated. The canopy deflection that was experienced during the Phase I Air Force evaluation flights in the higher supersonic speed ranges has been improved and is now satisfactory. Noxious fumes are always present when coming out of afterburning. These fumes were present for only a short time before clearing away. Fumes were not concentrated enough to cause eye smarting. For all practical purposes, the objectionable cockpit fumes that were present during the Air Force Phase I evaluation flights have been eliminated. Cockpit pressurization is not satisfactory when utilizing normal (5 psi) pressure differential. Small throttle adjustments affect cabin pressure differential, making it very uncomfortable to the pilot. This could possibly be attributed to a faulty pressure regulator. The heating and ventilation is considered adequate and distribution is satisfactory. No difficulties were experienced with the canopy fogging over at altitude or during rapid descents. The greater percentage of test flying was performed in the dry air of the Mojave Desert. The seat back angle is satisfactory for all phases of operation.

■ range mission

A combat radius mission was carried out in accordance with the following loading and flight plan as furnished by the contractor:

1 Loading

Fuel	13,150 lb at 6.53 lb/gal	=	2013 gal
Engine start gross weight		=	40,680 lb

2 Flight Plan

a Allow 5 minutes of normal rated power for warm up, taxi, take-off and acceleration to best climb speed.

Take-off in military power.

b Military power climb on course to optional cruise altitude of 34,100 feet.

c Cruise out climbing to 36,000 feet at speed for 99% maximum range.

d Military power climb on course to cruise ceiling of 44,500 feet.

e Combat at 44,500 feet.

15 minutes at military power, high speed.

5 minutes at maximum power acceleration.

f Descent to optional cruise altitude of 37,800 feet and cruise in, climbing to 39,600 feet at speed for 99% maximum range.

g Reserve, 10% of initial fuel loading.

The range mission profile is presented in Figs. 8 and 9, Appendix I. Cruise speeds used during the mission were those recommended by the contractor and were .02 Mach number higher than those obtained from the speed-power data. Intercompressor bleed valves were manually closed throughout the flight. The pilot was unable to light the afterburners at 42,000 feet during the combat portion of the mission. The right afterburner was lit at 41,000 feet but he was still unable to light the left afterburner at 39,000 feet. The combat operation of the Range Mission was discontinued at this point and cruise-in operation begun. The fuel flow data appearing on the plots was obtained by use of Potter fuel flow meters. The amount of fuel used and gross weights were calculated from fuel quantity gages which have an estimated collective accuracy of 200 pounds. The following table outlines the radius mission results:

	Fuel Used lb	Dist. Traveled Naut Air mi	Avg. True A/S kts	Time hrs	Total Elapsed Time hrs
A. Military power take-off and climb to cruise altitude of 33,500 feet.	1730	45.5	380	.117	.117
B. Cruise out climbing to 36,000 feet.	3180	380.5	477.5	.795	.912
C. Military power climb to cruise ceiling of 44,500 feet.	325	25	484	.052	.964
D. Combat — Military power.	1390	. .	. .	.267	1.231
E. Combat — Maximum power, afterburner wouldn't light.	600	. .	. .	.112	1.343
F. Descent to cruise altitude of 37,500 feet and cruise-in climbing to 40,000 feet.	4175	646	475.5	1.359	2.702
G. Reserve: over home base.	1765				
total:	13,150				

The cruise data obtained during the cruise-out operation at 35,000 feet resulted in higher power settings, higher fuel flows and a lower range than expected from level flight data. The cruise-in data at 40,000 feet was consistent with previously obtained level flight data. By making adjustments for the amount of fuel that would have been used during 5 minutes of afterburner operation and correlating range mission cruise data with speed-power data the following range would be realized:

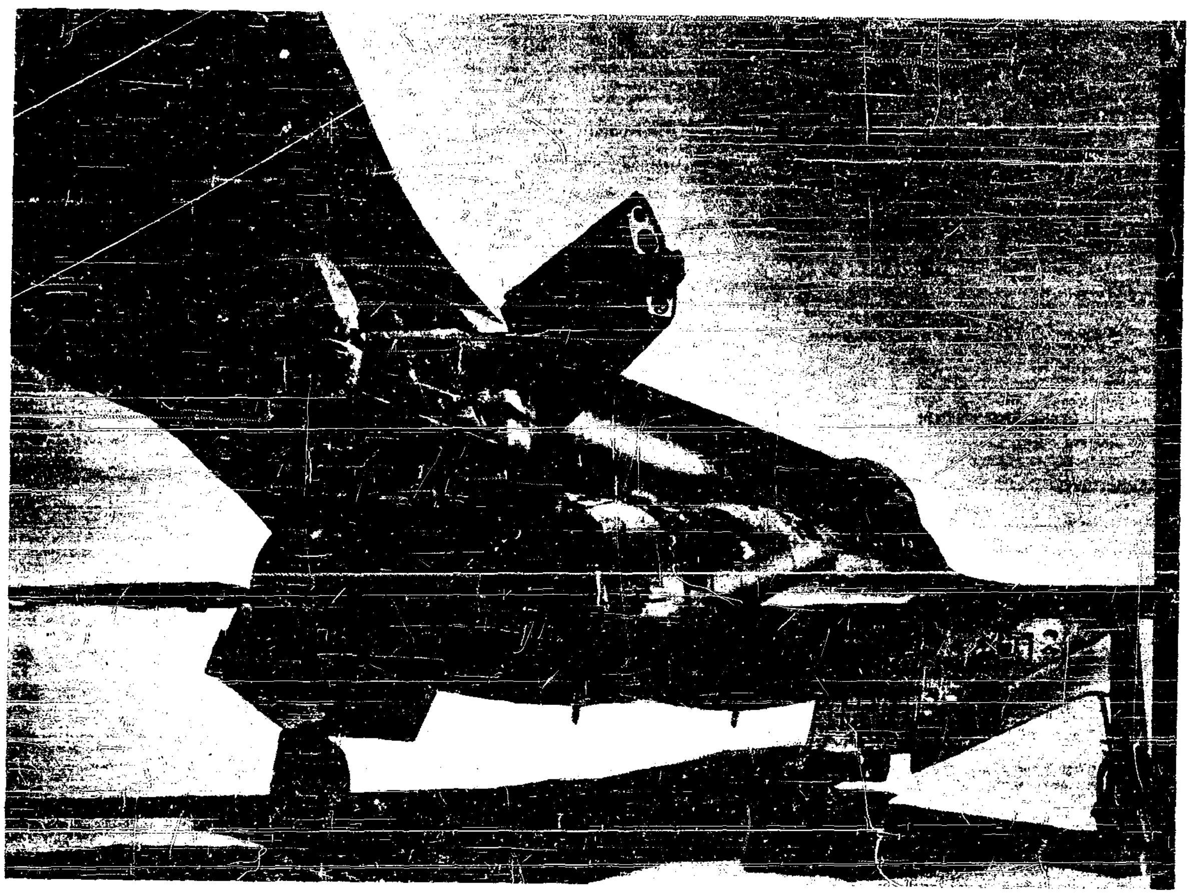
OPERATION	Fuel Used lb	Dist. Traveled NAM
A. Allowance for engine start, taxi, military power, T.O. and acceleration to best climb speed.	1070	0
B. Climb — military power to 33,500 ft	820	32
C. Cruise out	3420	437
D. Climb to cruise ceiling of 44,500 feet, military power	325	25
E. Combat — 15 min of military power 5 min of maximum power	3025	. . .
F. Cruise in	3175	494
G. Reserve	1315	. . .
total	13,150	radius of action 494 NAM

airspeed calibration

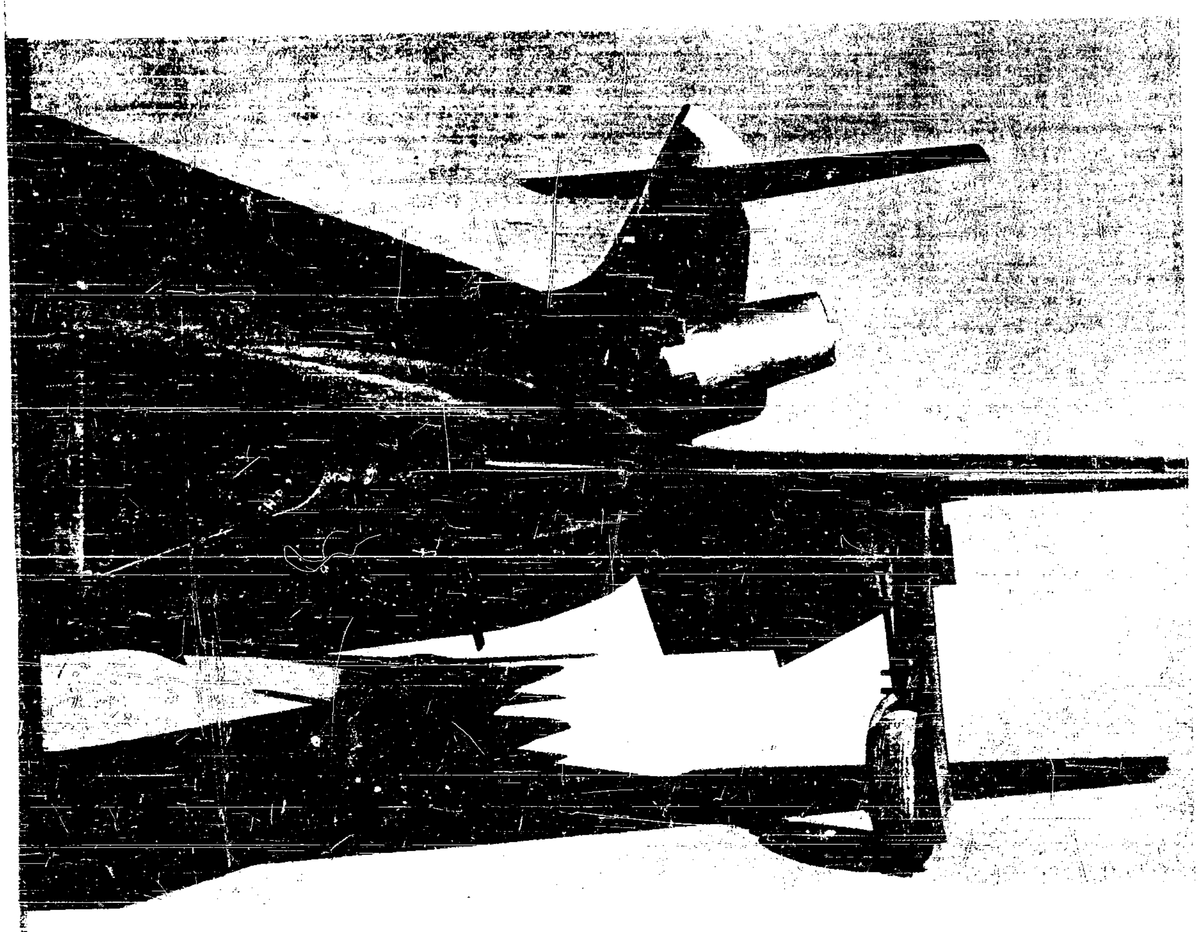
A test nose boom airspeed system and the ships standard airspeed-altimeter system were calibrated throughout the usable airspeed range by pacer aircraft and by accelerating and decelerating past the pacer. The systems were calibrated in the clean and landing configuration. The data is presented in Figures 18, 19 and 20, Appendix I.

normal approach and landing

The initial approach to a landing is made at an altitude of approximately 1000 feet above the runway. Indicated airspeed at break-off is between 300 and 350 knots. Speed brakes are extended after turning on the downwind leg in order to quickly decrease the airspeed to the gear down limit speed of 250 knots IAS. The speed brakes are normally closed



as the gear starts to extend. The base leg is flown at approximately 200 knots IAS and requires 75 to 80% rpm. to maintain airspeed and altitude. While turning on to "final" the landing flaps are lowered and the approach speed reduced sc as to roll out on "final" with 175-180 knots. The speed is further reduced to 165 knots for final approach. The power setting re·juired will be approximately 70% rpm. The rate of sink will be excessive if power is reduced to idle. Initial buffet will occur at approximately 155 knots IAS. The recommended speed for touchdown with 2000-3000 pounds of fuel remaining is 140-145 knots IAS. On one occasion by utilizing power the aircraft was flown down to a speed of



122 knots IAS before touchdown; however, buffet was excessive. The excessive buffet was felt to be caused by a disturbance at the wing root when flaps are extended. Buffet was present in no-flap landings, but to a much smaller degree.

Visibility throughout the final approach and landing is excellent. Immediately after touchdown the nose is lowered to the runway and upon contact the drag chute deployed. The drag chute is very effective and was successfully deployed for all landings during the Phase II tests. Braking after touchdown is good. If the buffet boundary can be reduced 15-20 knots, normal touchdown speeds can be reduced the same amount thereby permitting the aircraft to be safely operated from a shorter runway. The quality of the tires during the Phase II test was poor and the pilots were reluctant to use brakes for fear of a blowout.

Two performance landings to determine the distance required to clear a 50-foot obstacle were recorded on the AFFTC photogrid. The drag chute was deployed after touchdown and light braking used during the ground roll. Test data corrected to standard NACA atmosphere, sea level, no wind conditions is presented in the following table:

					Nose Boom		Ships System	
Gross Wt lb	Gr. Roll ft	Tot. Dist. over 50' – ft	TAS at 50' kts	TAS at TD kts	IAS at 50' kts	IAS at TD kts	IAS at 50' kts	IAS at TD kts
30,250	**5100**	**6360**	**161**	**147.5**	**163.5**	**145.5**	**161.5**	**146.5**
31,500	**5115**	**5410**	**161**	**150**	**161.5**	**145.5**	**159.5**	**145.5**

It is felt that if heavy braking could be used, a ground roll of 2500 to 3000 feet could be attained.

Several no-flap landings were made. The approach angle for no-flap landings is much flatter than with flaps. Final approach speed was 180 knots IAS and touchdown speed 155 knots IAS. Initial buffet occurs at 166 knots.

Cross wind landings were accomplished in gusty winds up to 35 knots. No great difficulties are experienced during the cross wind approach to a landing until buffet is encountered prior to touchdown. A straight path can easily be maintained by "crabbing" or dropping the up wind wing. Once buffet is encountered, the pilots tend to overcontrol the aircraft because of lateral control sensitivity. After touchdown the same problems exist in the landing ground roll that were brought out in the discussion on the cross wind take-off ground roll.

SINGLE ENGINE APPROACH AND LANDING

Data was obtained qualitatively for single engine approaches, landing and wave-offs in the event of an emergency. The figures represent pilot-observed data and are not corrected to standard day conditions. Flights were flown at altitudes above 2,270 feet.

Configuration	Single eng. power	Highest altitude at which level flight could be maintained
Gear down, flaps down, speed brakes out	Military	Not able to maintain level flight
Gear down, flaps up, speed brakes out	Military	6,000 feet
Gear down, flaps down, speed brakes in	Military	8,500 feet
Gear down, flaps up, speed brakes in	Military	10,500 feet

From the above data it can be seen that the best single engine approach configuration is with gear down, flaps up and speed brakes retracted. This configuration provides adequate excess power in the event of go-around. The approach speed should be approximately 15 knots higher than normal for a given amount of fuel remaining. The approach angle is relatively flat. Just before touchdown the flaps and dive brakes should be extended. The pilot must be certain that he has the runway "made" before extending the flaps and dive brakes. In the event of a go-around, utilizing military power, the aircraft will not maintain level flight much less climb with gear and flaps down and dive brakes extended. If a decision is made to go-around prior to extension of flaps and speed brakes, the aircraft will accelerate and climb nicely on single engine military power. The decision to go-around should be made early in the approach because once the normal rate of sink is established, approximately 250 feet of altitude will be lost after full throttle is applied (military power) and the aircraft accelerates into a climb.

dive and high speed flight

No extreme high angle dives were performed. One high speed dive was made from an altitude of 40,000 feet at an estimated dive angle of sixty degrees. The dive was terminated at 30,000 feet at an indicated Mach number of 1.35 with an abrupt constant 3 g symmetrical pull-out. The flying characteristics of the aircraft and the stabilizer effectiveness are excellent. With the yaw damper off there is a slight snaking tendency at approximately .94 and 1.30 indicated Mach numbers. This slight snaking tendency is not present with the yaw damper on. Caution must be exercised when making high g pullouts at low altitudes. The light stick forces and excellent stabilizer effectiveness may lead to overstressing the aircraft.

high altitude operation

The high altitude performance of the aircraft as previously mentioned, is seriously hampered by poor engine performance and compressor stalls under accelerated flight maneuvers. From a straight away climb after take-off with a full load of fuel, the F-101A cannot climb over 50,000 feet. To perform an intercept in this aircraft, the best procedure is to climb from take-off to 42,000 feet. At this altitude the aircraft is leveled off and allowed to accelerate to a speed of approximately 1.4 Mach number, reducing weight as fuel is consumed. A climbing turn to 48,000 feet can be made from this altitude, at 1.05 Mach number, holding 1.3 to 1.5 g. The climb can be continued to 50,000 feet at 1.02 Mach number.

A constant altitude, constant speed (1.02 Mach number) 1.2 g turn was made at 50,500 feet at a gross weight of 31,500 pounds. Temperatures were from 5° to 6°C colder than a standard NACA day.

night flight evaluation

The overall cockpit lighting and night visibility is generally good. However, there are three items which require improvement. The lighting and control of light intensity for the forward panel is considered adequate with the exception of the altimeter and airspeed indicators located on the extreme left portion of the forward panel. When the light intensity for the rest of the forward panel is properly adjusted, the light intensity for the altimeter and airspeed indicators is too low. When light intensity for these two instruments is adjusted properly, the light intensity for the rest of the panel instruments is too great. The holes for the various instrument adjustment knobs are too large for the knobs and emit too much light. This makes it difficult to read the instruments and also creates objectionable canopy glare and reflections. The console lighting and control of light intensity is considered excellent. The landing lights are focused too high to be effective for landing and create a milky effect in front of the pilot while on final approach thereby interfering with his depth perception.

TEST RESULTS stability and control

test configurations

The following configurations as per military specification MIL-F-8785 (ASG) are presented on the plots and referred to in the stability and control discussion:

Configuration: Co: Combat: Augmented power, airplane in combat configuration.

Configuration: CR: Cruise: Power for level flight at trim speed, flaps up, gear up.

Configuration: PA: Power Approach: Gear down, flaps down, speed brakes extended.

Configuration: L: Landing: Power off, gear down, flaps down, speed brakes in.

Configuration: TO: Take-off: Gear down, flaps down, take-off power.

Phase II stability tests were conducted at the normal cg loading. No control over cg location was exercised by ballasting or by fuel flow because the fuel selection was entirely automatic. The cg at engine start gross weight of 40,600 pounds was 31.4% MAC. The cg traveled aft 5.6% MAC with all internal fuel consumed.

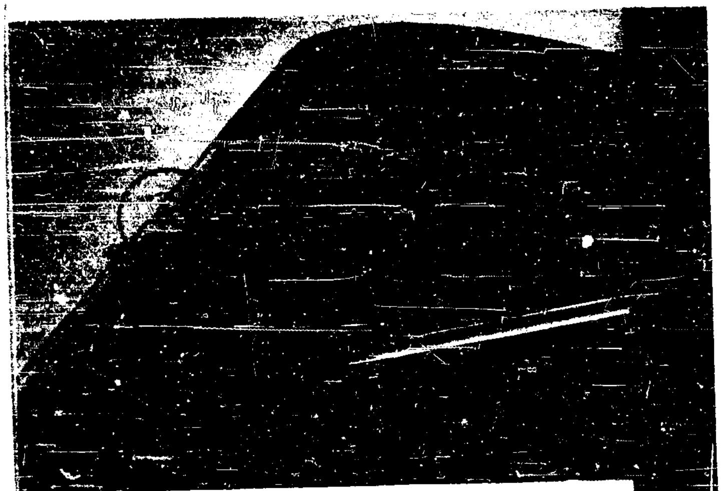
flight control system

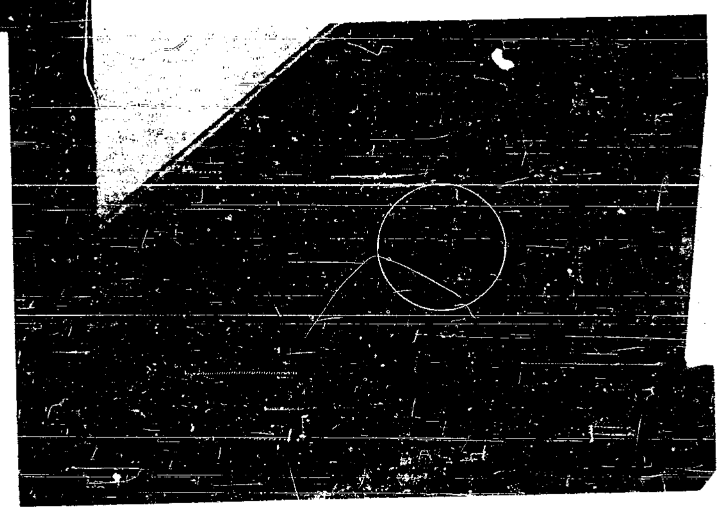
All primary controls are hydraulically powered and are irreversible in their action. Movement of the cockpit controls serves only to position valves, through mechanical linkage in such a manner that control surface movement is proportional to stick movement. The horizontal stabilizer and ailerons utilize power from both the primary and utility hydraulic systems. The rudder is operated by the primary system alone. To compensate for the irreversible control system, stick and rudder forces are provided by separate artificial feel systems. Each of the primary controls can be trimmed electrically.

Horizontal Stabilizer: The functions of the stabilizer and the elevator are combined into a single unit called the stabilator. The actuating unit consists of two cylinders with their pistons affixed to a common shaft. One cylinder is connected to the primary hydraulic system, the other to the utility hydraulic system. In normal operation both work simultaneously, but the horizontal stabilizer can be controlled by either, at reduced airspeeds, in the event of failure of the other.

"Feel" is induced into the stabilator control system by ram air pressures acting on the stabilator bellows. The bellows force is applied to variable balance assembly which is connected to the control stick through the control linkage. Stabilator bellows

forces vary with airspeed and air density. A viscous damper is incorporated in the feel system to provide a greater resistance to any abrupt or sudden stick movement and aid in preventing the airplane from being subjected to over-limit load factors. A bob-weight is also installed on the stick which will increase stick force with an increase in g load. During the original preliminary evaluation of the F-101A, the total pressure-sensing feel system pick-up for the inlet side of the diaphragm was a protruding pipe installed on the leading edge of the vertical fin (Fig. 6). Stick forces increased sharply at approximately 2.5 g indicating that at the limit load factor of 7.3 g, stick forces would be excessive. Prior to Phase II the pressure pick-up was relocated to approximately the middle of the left side of the vertical fin (Fig. 7). The pick-up was a flush duct type. With this location, stick forces both subsonic and supersonic were considered ideal. However, at the start of the Phase II flight program a longitudinal stick shake developed at an indicated Mach number of .95. The stick shake was alarming enough to





restrict the flight test program to speeds below 1.2 Mach number with the pick-up in this position. As a result the total pressure-sensing feel system pick-up was temporarily relocated to its original position on the leading edge of the vertical fin. The pick-up was a flush-tube type instead of a protruding pipe. This location gave somewhat the same longitudinal stick forces through the cruise and combat configurations but forces were lighter in the power approach and landing configuration. Although no quantitative data was obtained, qualitative data indicated that the force gradient in the power approach and landing configuration was low enough to be considered marginal.

Stabilator trim is accomplished through the use of the same mechanisms which provide artificial feel, namely the bellows force acting on the balance assembly. Longitudinal trim is actuated by use of the trim button on the top of the stick. The trim rate was considered to be a little too slow during gear and flap retraction after take-off. Because of the very light stick forces, however, no difficulty was experienced. It is felt that with higher longitudinal stick forces, a faster trim rate will be required.

Aileron: Ailerons are actuated, individually but simultaneously, by two tandem-type power cylinders using both primary and utility hydraulic power in the same manner as in stabilizer control. The connection between the stick and the power control valves in each actuating unit is a series of push rods and torque tubes. There is a spring rod incorporated in the linkage to each aileron so that in the event one aileron becomes immovable through battle damage or some other cause, the other can still be operated with a little additional pilot effort. Also, an additional spring cartridge is located at each aileron which will position the surface within ±5 degrees of neutral in the event linkage is shot away while the aileron is deflected.

A spring cartridge, one end of which is attached to the aileron linkage, supplies feel to this system. It is so designed that movement in either direction requires compression of the spring and consequently provides resistance to movement. Trim is provided by attaching the other end of this same cartridge to an electric actuator. When energized, the actuator moves the entire aileron linkage so that the neutral, or no-load position can actually be established with a small amount of aileron deflection in either direction. Aileron trim is considered unsatisfactory. Aileron trim is controlled by the same button on the top of the stick that controls longitudinal trim. The aileron trim response from the time the button is actuated is very slow and results in an overshoot of trim. In a static condition on the ground when the aileron trim is energized and released, the ailerons will overshoot for a five second period. It is very difficult to trim the aircraft laterally and to make small trim corrections due to the extreme lateral sensitivity.

Rudder: The rudder is operated by a single-cylinder power unit from the primary hydraulic system. The power cylinder servo valves are mechanically positioned by a single series of cables from the rudder pedals. In the event the primary hydraulic system becomes inoperative, power cylinder acts as a solid link in the cable system to permit rudder control by pilot effort. Fore and aft rudder pedal positions can be adjusted by the crank on the lower portion of the pedestal. Two rudder dampers are installed in the control system to counteract high-speed flutter tendencies.

The difference in the forces exerted on opposite sides of a piston within a hydraulic cylinder offers the resistance to rudder pedal movement necessary to produce the required feel. This cylinder joins the rudder linkage in such a manner that control movement in either direction tends to extend the cylinder. At slower airspeeds, full hydraulic system pressure is diverted to both sides of the piston, but due to the difference in area of the two sides, some resistance to pedal movement is provided. At an approximate speed of 285 knots, a speed switch is actuated which in turn operates a valve shutting off hydraulic pressure to the extension side of the piston. This means that full hydraulic system pressure then opposes rudder control movement, making it very difficult.

When the shift in hydraulic pressure occurs a definite rudder "kick" or force feed-back can be felt. When accelerating or decelerating through 285 knots, the rudder kick is noted and requires retrimming of the rudders. Throughout the speed range rudder forces are much too high.

Trim is accomplished by varying the neutral, or no-load position of the control linkage. This is made possible by an electric actuator in the feel system linkage which by extension and retraction moves the entire rudder control system proportionally. The actuator is energized by a three-position toggle switch on the left console which is spring-loaded to the "OFF" (center) position. Rudder trim is considered satisfactory.

The longitudinal, lateral and rudder control forces required to move the control surfaces throughout their deflection range were measured in a closed hangar. The static control forces are presented in Figures 29 through 33, Appendix I. The friction band is described by the width of the control force versus control deflection band described by arrows showing direction of application.

Control	Ground Static Friction Force	Maximum Allowable Force from MIL-F-8785(ASG)
Stabilizer	3.5	3
Aileron	1.5	2
Rudder (low feel)	27.	7
Rudder (high feel)	20.	7

stabilizer control effectiveness

Take-offs were performed at a gross weight of 40,400 pounds at a center of gravity position of 30.2 percent MAC. Stabilizer control during take-off is considered satisfactory. With the large amount of stabilizer control available it is believed that there will be adequate longitudinal control at the forward center of gravity limit of 26.5 percent MAC. Stick forces at nose gear lift-off were less than 10 pounds and are well under the maximum of 30 pounds specified by MIL-F-8785(ASG). Time histories of take-off are presented in Figures 34, 35 and 36, Appendix I.

Landing tests were flown to determine the effectiveness of the longitudinal control in holding the aircraft off the runway until maximum ground angle is reached. Landings were made at gross weights of 30,140 and 31,455 pounds, and at centers of gravity of 33.6 and 32.9 percent MAC. Stabilizer control effectiveness is considered excellent and stick forces are light. The 10-pound maximum recorded forces are well within the 35-pound MIL-F-8785(ASG) requirement. The light stick forces tend to make the pilot refrain from trimming the aircraft properly during approach. Although the light stick forces coupled with the slow longitudinal trim rate of the aircraft present no great problem, the pilot should exercise some care in trimming forces to the desired amount. Time histories of landings are presented in Figures 37 and 38, Appendix I.

static longitudinal stability

The static longitudinal stability in the combat and cruise configuration is positive, control is good, and forces are light throughout the greater portion of the speed envelope. There is a slight trim change in the transonic speed range (.94 to .96 Mach number) which results in reversal in stabilizer slope and stick force gradient with speed. The stick force and position reversals were hardly noticeable to the pilot. At supersonic speeds there is only a small change in the amount of stabilizer travel required for a large change in speed. Forces are light and the gradient is mild. This data is presented in Figures 40 through 43, Appendix I.

In the power approach configuration, the aircraft exhibits positive static stability; however, the light stick forces are more noticeable here than in other configurations. Aircraft response to control movement is good, but because of the light stick forces the pilot may have a slight tendency to overshoot the desired speed or attitude. Generally speaking, good static stability qualities tend to make the aircraft easy to fly and the longitudinal control during formation flying presents no problem. The data is presented in Figure 39, Appendix I.

dynamic longitudinal stability

The dynamic longitudinal stability characteristics were investigated throughout the speed range of the aircraft at altitudes of 12,000 and 36,000 feet. The tests were conducted with the controls free and with the pilot attempting to reduce the oscillations as quickly as possible. No control fixed tests were made because, with the irreversible control system, the results would be essentially the same as with the controls free. Tests were also made with both total-pressure feel systems installed. This data is presented in Figures 44 through 65, Appendix I.

The dynamic oscillations of normal acceleration damped to less than one-half amplitude in one cycle in all configurations with the exception of the cruise configuration at an altitude of 38,400 feet and calibrated airspeed of 278.5 knots. The magnitude of the residual oscillations exceeded +.02 g in almost all cases; however, the angular attitude was not objectionable to the pilot. The motion of the stabilizer following release was essentially deadbeat. Pilot induced oscillations did not result from the pilot's efforts to damp the residual oscillation as quickly as possible.

With the total pressure artificial feel system pick-up located on the leading edge of the vertical fin, the induced oscillations took a slightly longer time to damp out. This is more pronounced in subsonic speeds than in the supersonic flight regime, and was most apparent in the power approach configuration. The light stick forces and damping characteristics at cruise speeds may require a pitch damper to aid in tracking. A check on the tracking characteristics of the aircraft should be made as soon as possible.

maneuvering flight characteristics

The maneuvering capabilities of the F-101A in the supersonic flight range are slightly better than those of the F-100, and are less in the subsonic flight range. The aircraft is seriously restricted in its ability to maneuver at altitude and its capability as an air superiority fighter compromised because of compressor stalls which result from positive g loads. The problem is most pronounced in subsonic maneuvering. Compressor stalls will usually occur in positive normal acceleration turns of 2.5 to 3 g at altitudes of 45,000 to 50,000 feet while operating in afterburning at subsonic speeds. During supersonic maneuvering the problem also exists in normal acceleration turns of 3 to 4 g. Compressor stalls occur in the inboard engine and usually result in afterburner blow-out followed by rapid engine stalls until g forces are relaxed. If compressor stalls are encountered at any time during combat, the pilot would immediately be placed on the defensive, have to break off combat, relax g forces, lose altitude and pick up airspeed in order to accelerate through the stalls.

The maneuvering flight capabilities of the aircraft in the supersonic flight range, discounting compressor stalls resulting from positive g loads, are considered excellent. At altitudes up to 40,000 feet at speeds varying from 1.3 Mach number to V_{max}, 4 g could be attained easily. The limit load factors for the Phase II test were +4 g to 0.5 g. The stick force gradient is low enough to indicate that the stick forces will be within limits at the design limit-load factor of 7.33 g. The ability to perform constant altitude, constant airspeed turns at supersonic speeds at an altitude of 45,000 feet is superior to any other fighter aircraft tested at the Flight Test Center. The following data was obtained with the pilot executing constant altitude and constant airspeed turns:

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
44,260	401.5	1.349	1.86	33,390
44,540	385.0	1.306	2.13	32,330
34,380	486.0	1.326	2.56	33,290

Control forces in steady accelerated flight were obtained in the combat, cruise and power approach configurations. Tests were conducted by placing the aircraft in steady turning flight to obtain stabilized conditions of positive normal acceleration, stick force and speed, while allowing the altitude to vary as little as possible. The stick force gradient in all configurations was within the limits as specified by MIL-F-8785 (ASG). The data is presented in Figures 66 through 74, Appendix I and is summarized in the following table. The limit load factor of 7.33 g was used to compute maximum and minimum force gradients.

STABILIZER CONTROL FORCE GRADIENT

Configuration		CG % MAC	Average Force Gradient lb/g	Maximum g Attainable g	Maximum Force Gradient MIL-F-8785(ASG) lb/g	Minimum Force Gradient MIL-F-8785(ASG) lb/g
CRUISE	CAS 282 kts alt. 37,400 ft wt. 31,250 lb	34.7	7.2*	1.7	8.85	3.32
CRUISE	CAS 294 kts alt. 35,900 ft wt. 31,250 lb	34.7	8.1*	1.9	8.85	3.32
COMBAT	CAS 474.5 kts alt. 35,000 ft wt. 32,050 lb	33.9	7.0*	..	8.85	3.32
COMBAT	CAS 508.5 kts alt. 36,500 ft wt. 33,780 lb	34.0	7.0**	..	8.85	3.32
CRUISE	CAS 418 kts alt. 13,800 ft wt. 37,580 lb	32.9	3.53*	3.5	8.85	3.32
POWER APPROACH	CAS 193 kts alt. 12,700 ft wt. 31,750 lb	32.43	5.5*	1.55	8.85	3.32
CRUISE	CAS 483 kts alt. 14,700 ft wt. 38,000 lb	31.4	4.5**	3.6	8.85	3.32

* Total pressure sensing artificial feel system pick-up located on the leading edge of vertical fin.

** Total pressure sensing artificial feel system pick-up located on the left side of vertical fin.

The stick force gradients in the cruise and combat configurations were similar with either total pressure sensing artificial feel system pick-up. Although no quantitative data was obtained, the pilot felt that the stick force gradient with the total pressure sensing feel system located on the vertical fin was too low.

High speed dives and abrupt pull-outs were flown to demonstrate that the stick force gradient during abrupt pull-outs is equal to or exceeds that obtained during steady accelerations. This data is presented in Figures 75 and 76, Appendix I. In both dives and pull-outs the stick force gradients were higher than those recorded in steady accelerations.

longitudinal trim changes

The longitudinal trim changes caused by changes in power, gear operation, etc., encountered under normal flight operation were investigated. Trim changes are small and are easily controllable by light stabilizer control forces. Flap extension or retraction produces the most noticeable single longitudinal trim change. A simulated wave-off from the approach configuration, trim speed 180 knots IAS, was made. The wave-off was initiated by applying maximum power then retracting landing gear, flaps and speed brakes. The resulting trim changes were easily handled because of the light longitudinal control forces and effective stabilator control.

In the approach configuration, 180 knots was the lowest indicated airspeed the aircraft could be trimmed to and maintain hands-off level flight. At 180 knots, IAS, the up stabilator trim was gone due to the forward movement of the stick that results from speed brake extension (see Speed Brake Operation).

The following table presents the longitudinal trim change conditions and the peak longitudinal control forces required subsequent to the pilot action initiating the configuration change. MIL-F-8785 (ASG) specifies that the magnitude of the peak longitudinal forces shall not exceed 10 pounds for a period of 5 seconds following the configuration change.

initial trim conditions

Condition No.	Altitude ft	CAS kts	Gear	Flaps	Power	Config. Change	Parameter to be held Constant	Peak Longitudinal Force — lb
1	14,760	232	up	up	P.L.F.*	gear dn	alt.	3.6 pull
2	14,970	236.5	dn	up	P.L.F.	flaps dn	alt.	4.6 pull
3	14,380	233.5	dn	dn	P.L.F.	idle power	speed	5.6 pull
4	15,420	173.5	dn	dn	P.L.F.	take-off power	alt.	13.4 push
5	9,210	163.5	dn	dn	take-off	gear up	rate of climb	***
6	9,210	163.5	up	dn	take-off	flaps up	rate of climb	7.5 push***
7	35,100	328.5	up	up	MRP**	idle power	alt.	3.0 pull
8	35,350	330	up	up	MRP	extend dive brake	point of aim, alt.	7.5 pull
9	11,280	433	up	up	P.L.F.	extend dive brake	alt.	5.6 push
10	11,660	210.5	dn	dn	P.L.F.	extend dive brake	speed	4.2 push
11	35,240	338.5	up	up	MRP	augmented power	alt.	1.3 pull

* Power for level flight at specified conditions.

** Military rated power.

*** The peak longitudinal force listed for condition No. 6 was the peak force recorded for condition No.'s 5 and 6 run in sequence with one another.

speed brake operation

The effectiveness of the F-101A speed brake was cross checked against speed brake effectiveness of an F-86F. The results indicated the F-101A speed brake is more effective. Reduction of the speed brake travel to fifty degrees (approximately ¾ open) has eliminated the objectionable buffet reported during the Air Force Phase I evaluation flights. There is very little objectionable trim change when the speed brakes are open or closed. When the speed brakes are opened, the control stick will move forward to such a degree (one to two inches at 180 knots IAS) that only a normal acceleration of +0.5 g is induced. The aircraft gains approximately 500 feet in altitude with hands off the stick. The force required to maintain constant altitude is light. The reverse movement of the stick occurs when the speed brakes are closed. This characteristic increases the stabilizer travel available and provides a degree of automatic trimming. Time histories of speed brake opening and closing are presented in Figures 77 through 81, Appendix I.

dynamic lateral and directional stability

Dynamic lateral and directional stability tests were flown to determine the response of the aircraft to either lateral or directional disturbance. Tests were flown at altitudes of 11,500 to 14,000 feet in the power approach, cruise and combat configurations and at 38,000 feet in the cruise configuration. Dynamic lateral tests were conducted with the controls free and with the pilot attempting to reduce the oscillations as rapidly as possible. Tests were also conducted with the yaw dampers "Off" and "On". Dynamic directional tests were conducted with the controls free and the yaw damper "Off" and "On".

The dynamic lateral directional stability with the yaw damper operative and working properly is excellent. The directional oscillations produced by abruptly displacing the rudder are damped within one cycle. Experience during the Phase II tests indicates that proper operation of the yaw damper may be a source of difficulty. Twice during the program a malfunctioning rate gyro had to be replaced.

The dynamic lateral and directional stability with the yaw damper off is unsatisfactory throughout the entire speed range. The time for the damping of the lateral-directional oscillations is excessive. In smooth air when the rudder is displaced, directional oscillations are not completely damped after three cycles and show no tendency to damp at all under slightly turbulent air conditions. Without the use of the yaw damper the lateral-directional stability characteristics of the F-101A detract from the airplane's capabilities as a gun platform. The multi-purpose functions of the F-101A make a properly working yaw damper a necessity.

Damping of lateral disturbances was executed by leaving the controls free (which for an irreversible system is the same as a control fixed test) and also with the pilot attempting to damp out the disturbance as quickly as possible. The sensitivity of the aileron, especially in the speed range from .75 to .9 Mach number, allowed the pilot to overcontrol and induce a violent lateral oscillation. A graphic example of this is shown in Figure 82, Appendix I. Lateral disturbances were most quickly damped by releasing the control.

static directional stability

Static directional stability of the F-101A was quantitatively investigated in the power approach configuraton at 12,500 feet and in the cruise configuration at 16,000 feet with and without the yaw damper operative. Static directional stability was also investigated qualitatively throughout the speed range of the aircraft but no quantitative data was obtained because of instrumentation malfunctioning and unreliability. The static directional stability is positive. No differences were discernible with or without the yaw damper. Only small angles of sideslip are attainable because of the high rudder forces. The results of this test are presented in Figures 119 through 122, Appendix I.

For single engine operation, or single engine afterburner operation, directional control cannot be maintained without the use of rudder trim because of the high rudder forces. This deficiency could severely hamper target tracking during combat.

lateral control

The lateral control of the F-101A is unacceptable and is considered to be one of the serious deficiencies of the aircraft. Throughout the entire speed range of the aircraft the ailerons are too sensitive. This sensitivity is most pronounced in the speed range from .75 to .90 Mach number. In this speed range, the ailerons are too effective for a given stick dis-

placement. As was brought out in the Dynamic lateral Stability discussion it is very easy for the pilot to induce violent lateral oscillations. This is especially true in turbulent air. Releasing the stick will stop the oscillation. The cruise speeds of the aircraft are within the .75 to .90 Mach number range and for all practical purposes the aircraft will perform its assigned mis... flying to and from the target areas in this speed range. Formation flying will become difficult and instrument flying a hazard. The lateral sensitivity within the speed range from .90 Mach number to V_{max} does not present any safety of flight problem; however, the lateral control can be defined as marginal to satisfactory. In the speed range below .75 Mach number the lateral control, while not as sensitive as in the cruise range, is still unsatisfactory. At these lower speeds the most difficulty will be encountered in gusty winds from the initial approach to the down wind leg of the landing pattern. The sensitivity of the lateral control will cause the pilot to overcontrol and an exceptionally large pattern will result.

Aileron rolls were performed in cruise and combat configurations at 35,000 feet and in the cruise and power approach configuration at 16,000 feet. No full deflection aileron rolls were performed. Aileron deflections for the Phase II tests were restricted to 10 degrees up to 500 knots IAS and to five degrees of aileron deflection above 500 knots IAS. The limitation on aileron angle at high speeds was made pending the completion of static tests. Rolls were restricted to 45 degrees of bank until the inertia coupling problem could be fully investigated by the contractor. All rolls were performed with the yaw damper "on" and repeated with the yaw damper "off". The most noticeable amount of adverse yaw for the aileron rolls performed occurred at 38,000 feet at .84 Mach number with the yaw damper off. The yaw build-up was constant. It is very difficult to overcome adverse yaw because of the high rudder forces. Aileron rolls performed at 36,000 feet and 1.35 Mach number indicated very little adverse yaw. The data for the maximum allowable aileron rolls is presented in Figures 126 through 141, Appendix I, while a summary is presented in Figures 123 through 125, Appendix I. Although no full-deflection aileron rolls were performed it appears by extrapolation of the available data that the maximum helix angle (pb/2v) of .09 will be met for $1.1V_{S_G}$ and minimum combat speed as defined by MIL-F-8785(ASG).

adverse yaw

Adverse yaw tests were conducted at 35,000 feet in the cruise and combat configuration. Tests were performed with the yaw damper on and off. The aircraft was trimmed in a steady 45 degree bank and then maximum allowable aileron deflection was applied with the rudder pedals free. The maneuver was repeated with rudder coordinated. The angle of sideslip developed during these rolls appeared to be less than three degrees, but the instrumentation used to indicate sideslip was unreliable and gave erratic readings. No differences between the sideslip angles developed with the yaw damper on or off were apparent with the small angles of sideslip developed. The adverse yaw data is presented in Figures 142 through 145, Appendix I.

stall characteristics

Approaches to a stall were performed at 16,000 and 35,000 feet in the clean and landing configurations. No complete stalls were performed as the contractor has not yet investigated this region. When conducting approaches to a stall in the cruise configuration (clean airplane), at 35,000 feet and a gross weight of 32,400 pounds, the aircraft began to buffet at approximately 225 knots IAS. The stall approach was continued on down to 185 knots IAS at which point the left engine compressor stalled. The onset of initial buffet is delayed at lower altitudes and lighter weights. At an altitude of 16,000 feet and a gross weight of 32,750 pounds (clean airplane), the initial buffet was encountered at 180 knots IAS. In the landing configuration at 16,000 feet, initial buffet occurs at approximately 170 knots IAS and at 2400 feet initial buffet occurs at 155 knots IAS. Stall approaches were continued down to approximately 125 knots IAS, where they were discontinued because of severe buffet and the extreme lateral sensitivity of the aircraft. Adequate lateral control is present at the initial buffet, and touchdown speeds should be kept just within the initial buffet boundary; i.e., approximately 140-145 knots IAS. Because of the lateral control sensitivity in all configurations it is impossible to keep from overcontrolling and to prevent the aircraft from dropping off on a wing. It is the pilot's opinion that if it were not for the stall being aggravated by the lateral sensitivity, the aircraft would stall straight ahead. Stall recovery is made by releasing back pressure on the stick and per-

mitting airspeed to increase. Buffet is present during the stall recovery up to the point where it was initially encountered. Power-off stall recovery can be accomplished with approximately 5000 feet loss of altitude. By advancing the throttle to full power, stall recovery can easily be made with approximately 2000 feet loss of altitude.

Accelerated stalls are considered satisfactory inasmuch as aircraft buffet is similar to that found in other swept wing aircraft; however, the buffet boundary is small and there is very little buffet before a slight pitchup and wing drop occurs. Wing drop may occur in either direction. Stall recovery is made by releasing stick back pressure. Accelerated stall approaches were discontinued when engine compressor stalls occurred. At 10,000 feet in the cruise configuration and power for V_{max} range (.77 Mach number), engine compressor stalls occurred at normal acceleration of 2.8 g. At 40,000 feet in the cruise configuration and power for V_{max} range (.82 Mach number), engine compressor stalls occurred at normal acceleration of 3 to 4 g. In the landing configuration at 10,000 feet and 230 knots IAS, compressor stalls occurred at 1.8 g. Time histories of stall approaches are presented in Figures 146 through 153, Appendix I.

CONCLUSIONS

The F-101A airplane is unacceptable for service use because of its comparatively poor high altitude performance, restricted maneuverability due to engine compressor stalls under accelerated flight maneuvers, and lateral control sensitivity.

The rate of climb, maximum level flight speeds and acceleration characteristics are excellent. The range capabilities of the F-101A with the present bleed valve schedule are 15% less than the predicted values. The range can be increased by 6.5% however, by lowering the engine bleed valve schedule to insure closed bleed valve operation during cruise conditions.

The lateral control handling characteristics are unsatisfactory. The lateral control is too sensitive throughout the entire speed range. This is most pronounced in the cruise region making formation flying difficult and instrument flying a hazard.

The longitudinal control handling characteristics with the artificial feel total-pressure flush tube pick-up located on the leading edge of the vertical fin are good throughout the entire speed range with the exception of approach speeds where the light stick forces are marginal. The longitudinal control handling characteristics with the pick-up located on the left side of the vertical fin are satisfactory at approach speeds but unacceptable at high speeds because of a longitudinal stick shake existing at .95 Mach number and above.

The dynamic lateral-directional stability with the yaw damper working properly is excellent, but is unsatisfactory with the yaw damper inoperative. Directional control was adequate for normal flight operation; however, for single engine operation or single engine afterburner operation rudder forces are too high to maintain directional control without utilizing rudder trim. Rudder forces are excessive for all taxiing, ground handling and flight maneuvering.

The maneuvering flight capabilities of the F-101A in the supersonic speed range up to altitudes of 45,000 feet are excellent. Maneuvering flight capabilities in the subsonic range are less than those of the F-100. Maneuvering flight is seriously affected at all altitudes in the subsonic speed range by engine compressor stalls under positive loads of 2.5 to 4 g. This deficiency reduces tactical effectiveness of the aircraft as an air superiority fighter. This problem also exists during supersonic maneuvering at altitudes above 35,000 feet under 3 to 4 g loads. To successfully maneuver above 50,000 feet the F-101A will have to maintain supersonic flight.

Engine performance is considered unsatisfactory because of the previously mentioned engine compressor stalls, failure of the afterburner nozzles to remain closed during normal non-afterburning operation, erratic bleed valve operation and inability to relight the afterburner at 45,000 feet.

The F-101A can be developed into a good multi-purpose fighter by correcting the lateral control sensitivity, eliminating engine compressor stalls encountered in accelerated flight maneuvers and improving the high altitude performance.

RECOMMENDATIONS

The following items make the F-101A aircraft unsatisfactory for service use. It is recommended that a study be made and action taken to:

Increase the altitude capabilities of the aircraft.

Improve the lateral control by decreasing the lateral control sensitivity throughout the speed range.

Eliminate the engine compressor stalls encountered under accelerated flight maneuvers.

It is recommended that the following improvements and changes be incorporated in production aircraft as soon as possible:

The erratic engine bleed valve operation be improved.

The afterburner nozzles be redesigned to insure proper closing and to prevent them from drifting open while in flight.

The afterburner blowouts encountered above 50,000 feet and during accelerated flight conditions be eliminated.

The rudder forces be decreased at least 50% for both ground and air operations.

Throttle quadrant:

The throttle handgrips be reduced in size.

The spring in the speed brake opening switch be eliminated and the switch be relocated to conform with natural thumb operation.

A push-type microphone button be installed on the throttle to be utilized in conjunction with a two-position antennae selector switch located elsewhere in the cockpit.

A minimum detent or latch be installed on the throttle quadrant for afterburner operation.

The throttle friction lever be relocated outboard of the left throttle.

The nose wheel steering "engage" button be moved to the stick grip.

A more practical control for the take-off "AUX" position be incorporated. (A single push-type button that would engage both take-off locks would be sufficient.)

The nose wheel centering characteristics be improved.

The canopy electrical actuation be redesigned to enable the pilot to open or close the canopy at all times regardless of nose gear position.

The consoles be elevated approximately three inches and tilted inward an additional five degrees.

The hydraulic pressure indicating system be modified to indicated failure of individual hydraulic pumps.

The stabilator, rudder and aileron trim lights be incorporated into a single trim light.

The speed brake "open" warning light be relocated to the forward panel so that it is in line with the pilot's normal line of vision.

The noise level of the landing gear warning buzzer be reduced by approximately one-third.

The emergency wheel brake control be relocated to the left side of the panel.

To provide for quick and accurate determination of trim conditions, consideration should be given to interchanging the position of the needle and ball instrument with the rate of climb instrument.

The cockpit pressurization be developed to provide satisfactory service when utilizing normal pressure differential (5 psi).

Yaw dampers be installed on all production aircraft.

The overshoot in the lateral trim be eliminated.

The longitudinal trim rate be increased approximately 25%.

The clanking in the nose wheel well during taxi operations be eliminated.

For night operations:

> The distribution of the forward panel lighting be improved.
>
> The objectionable canopy glare and reflections be eliminated.
>
> The landing lights be readjusted downward.

The provisions for emergency landing flap extension be removed.

A yaw string be installed on all production aircraft.

Anti-icing provisions be installed on the inlet ducts of the airframe.

A support be incorporated in the seat cushion for the pilot's parachute.

It is recommended that a study be initiated to determine means of:

Improving the engine inlet ducting to eliminate engine compressor stalls.

Improving the reliability of the rate gyro in the yaw damper.

Eliminating the noxious fumes which are present from time to time, especially shortly after coming out of afterburning.

Eliminating or reducing the aggravating airframe buffet that occurs at approximately .90 indicated Mach number under unaccelerated flight conditions.

Decreasing the wide buffet boundary encountered in the landing configuration prior to stall.

Eliminating the rudder "kick" at 285 knots IAS caused by the change over in the rudder force feel system.

Improving and prolonging tire life.

APPENDIX I

TABLE OF CONTENTS

performance curves

stability curves

■ introduction

This section briefly summaries the methods of data reduction used in analyzing the test data.

■ references

The following publications will be referenced in the discussion on data analysis:

1. "Flight Test Engineering Manual," AF Technical Report No. 6273.
2. "Pressure Altitude Method of Flight Test Data Reduction," AMC Memorandum Report No. TSFTE-2060.
3. "A Method of Determining Delta Rate-of-Climb for Turbo-jet Powered Aircraft," AMC Memorandum Report No. MCRFT-2157.
4. "Standardization of Take-off Performance Measurements," AFFTC Technical Note R-12.
5. "Specification No. A-1665, J57-P-13 Turbo-jet Engine," Pratt and Whitney Aircraft, dated 15 October, 1952.
6. Preliminary Operating Instructions XJ57-P-13 and YJ57-P-13 Engines, dated 1 January 1954 revised 1 March 1955.
7. "Memo. No. EAFB-365," McDonnell Aircraft Corporation, dated 25 March, 1955.
8. "Model F-101A Summary of Estimated Aerodynamic Data, MAC Report No. 3683," McDonnell Aircraft Corporation, dated 16 September 1954.

■ data analysis methods

Take-offs: Take-off data was obtained for maximum power by use of the AFFTC photogrid facility. Test ground and air distances and speed were obtained from time history plots. This data was then reduced to standard day, sea level, no-wind conditions by methods outlined in references No. 1 and 4. The standard day thrust value of 11,920 pounds per engine was used for brake release. This value was arrived at as the thrust that would be obtained with the engine trimmed to the tail pipe exhaust gas pressure (P_{t7}) that would fall in the center of the trim tolerance ($\pm 0.5''$Hg) on trim curves from reference No. 6 and 7, and for an average over-shoot condition of 1.5″Hg. The thrust figures were determined from thrust calibrations of the tail pipe pressure probes made on the AFFTC Universal Thrust Stand. Thrust calibrations are presented in Figures 21 through 26, Appendix I.

Climb: All climb data was reduced to the rate of climb that would have been obtained under standard NACA day atmospheric conditions using maximum or military power as the case may be, for climb speeds used during the test program. The methods used for data reduction are outlined in References No. 1 and 3. Thrust data for these corrections was obtained from Reference No. 5. The derivation of the method of correction is outlined in Reference No. 3. A plot of F_n/δ_{t2} vs. $\%N_2/\sqrt{T_a}$ was made from reference No. 5, where $100\%N_2$ was equal to 9975 rpm. This plot was worked up in terms of $\%N_2$ rpm because each engine delivers rated thrust at a different rpm (N_2). It was assumed then that each engine would deliver roughly the same thrust at the same value of $\%N_2$ rpm. Since the fuel control unit of the J57-P-13G engine maintains a predetermined engine rpm (N_2) with respect to compressor inlet temperature it was necessary to determine this schedule. The data appears in the form of a rpm-bias curve in Figures 15, 16, and 17, Appendix I. Data from military and maximum power climbs and level flight

do not form a single curve because of a temperature lag that results from high rates of climb. The rate of climb for military power climbs was corrected to the rpm that would have been obtained on a standard day at the test day Mach numbers. The rpm-inlet temperature schedule for military power climbs was determined from Figure 16, Appendix I. Standard day%/oN$_2$ rpm are presented in Figure 2. One hundred percent rpm is that rpm obtained with the engine trimmed to the tail pipe exhaust gas pressure (P_{t7}) that would give rated thrust at sea level and a compressor inlet temperature of 15°C.

Thrust correction for maximum power climbs were also determined from reference No. 5. However, because there were no afterburning estimated curves of thrust versus true speed for parameters of engine rpm available, thrust corrections were made by reference to the curves of Estimated Effect of Ambient Temperature on Thrust by use of the following relationship:

$$\Delta F_n = \left[F_{n_s} - \left(\frac{F_{n_t}}{F_n \text{ at } 59° \text{ F}} \Big/ \frac{F_{n_s}}{F_n \text{ at } 59° \text{ F}} \right) F_{n_s} \right] = \left[F_{n_s} - \left(\frac{F_{n_t} \times F_{n_s}}{F_{n_s}} \right) \right]$$

where:

$\frac{F_{n_t}}{F_n \text{ at } 59° \text{ F}}$ = % net thrust at test day temperature

$\frac{F_{n_s}}{F_n \text{ at } 59° \text{ F}}$ = % net thrust at standard day temperature

F_{n_s} = standard day net thrust at test day Mach number, lb

F_{n_t} = test day net thrust at test day Mach number, lb

ΔF_n = net thrust on a standard day minus net thrust on a test day, lb

Standard weights and standard rates of climb were then determined from standard day fuel flow, times to climb at test weight and the weight correction equation presented in Reference No. 1.

Standard day exhaust gas temperatures were determined from a working plot of:

$$T_{t7_t} \,/\, \theta_{t2_t} \text{ vs. } N_{2_t} \,/\, \sqrt{\theta_{t2_t}}$$

by entering the plot at the actual test point and moving parallel to the curve to the values of standard day corrected rpm, $N_{2_s} / \sqrt{\theta_{t2_s}}$ and reading the values for standard day corrected exhaust gas temperature, $T_{t7_s} \,/\, \theta_{t2_s}$

Standard day exhaust gas temperature is then:

$$T_{t7_s} = \left(T_{t7_s} \,/\, \theta_{t2_s} \right) \times \theta_{t2_s}$$

Corrections to fuel flow were made in a similar manner from working plots of:

$$W_{f_t} \Big/ \delta_{t_{2_t}} \sqrt{\theta_{t_{2_t}}} \text{ vs. } N_{2_t} \Big/ \sqrt{\theta_{t_{2_t}}}$$

where:

N_2 = rotational speed of high pressure rotor, rpm

T_{t_7} = exhaust gas temperature, °K

W_f = engine fuel flow, lb/hr

θ_{t_2} = T_{t_2}/T_{SL} inlet temperature ratio

δ_{t_2} = P_{t_2}/P_{SL} inlet pressure ratio

T = Temperature, °K

P = Pressure, "Hg

The subscripts t and s refer to test and standard conditions. The subscript t_2 refers to total engine inlet conditions.

Maximum Speeds: A correction factor of .01 Mach number change for a 1°C change in ambient air temperature was applied to test day supersonic Mach numbers to obtain standard day Mach numbers attainable. This correction factor was derived from experience with other supersonic aircraft. The limited flight time for the Phase II test program prevented gathering enough supersonic data to determine a more exact correction that could be applied to the F 101A. It is believed that this correction may be a little too low. Weight corrections were made with the aid of estimated data from Reference No. 8. To determine maximum or military rated rpm for standard day conditions it was necessary to estimate the maximum attainable Mach number under standard day conditions which in turn would determine the standard day compressor inlet temperatures. The standard day maximum and military rpm could then be determined from the level flight rpm-bias curve. The actual test data was used in going from test day rpm-temperature to standard day rpm-temperature.

Cruise Data: All level flight data was obtained in stabilized level flight and by flying at a constant weight/pressure ratio (W/δ_a) to minimize weight corrections. The methods of data reduction are outlined in Reference No. 1. Weight corrections, for the most part, were small and were made by applying the correction factor: $\Delta N/\sqrt{\theta_a}/\Delta W/\delta_a$ as outlined in Reference No. 1. Fuel consumption data was corrected to the fuel consumption that would be realized under standard day conditions and standard W/δ_a, assuming a constant Mach number from test to standard day.

Landings: Performance landings to determine the distance required to clear a fifty-foot obstacle and come to a stop were recorded on the AFFTC photogrid. The two landings performed were not brought to a complete stop and it was necessary to make a time history plot of velocity from the time history plots of test ground and air distance to determine the airplane's deceleration along its ground roll. Test day data was corrected to standard NACA atmosphere, sea level, no wind conditions by equations presented in Reference No. 1.

Airspeed and Altimeter Calibration: The test nose boom airspeed system and the ship's airspeed system were calibrated in the clean and landing configurations by pacer aircraft up to the maximum speeds attainable by the pacer and through

the "jump" region by accelerating and decelerating past the pacer. This latter method was carried out by the F-101A flying on a contrail formed by the pacer flying at a constant airspeed and constant altitude. A plot of $\Delta P/q_{ic}$ vs M_{ic} was made from this data and correlated with the pacer calibration to obtain the dash line shown in Figure 18, Appendix I.

Temperature Probe Calibration: The variation of indicated free air temperature with airspeed was found from the pacer calibration. The temperature probe recovery factor, K, evaluated from the test data, was found to be approximately .94. The expression for determining the probe recovery factor is:

$$K = \left(\frac{T_{ic}}{T_a} - 1 \right) 5/M^2$$

The ambient air temperatures used for climb data reduction were determined from radiosonde data furnished by the Edwards Air Force Base Weather Station. This data is presented in Appendix III. Indicated free air temperatures recorded during the check climbs showed a temperature lag that varied with the rate of climb and climb speed. Indicated free air temperatures recorded during stabilized level flight were in agreement with radiosonde data. Compressor inlet temperatures were calculated from ambient air temperatures for a temperature recovery factor of K equal to 1.0

Static Thrust Calibration: Static thrust calibrations were made on each engine operating alone in afterburning and non-afterburning and with both engines operating together in non-afterburning. The methods of data reduction are outlined in Reference No. 1. Static thrust calibrations are presented in Figures 21 through 24, Appendix I. The exhaust nozzle pressure probes (P_{t7}) consisted of four probes manifolded together. The pressure readings were recorded on a single sensitive manifold pressure gage. The probe calibration data is presented in Figures 25 and 26 of this Appendix.

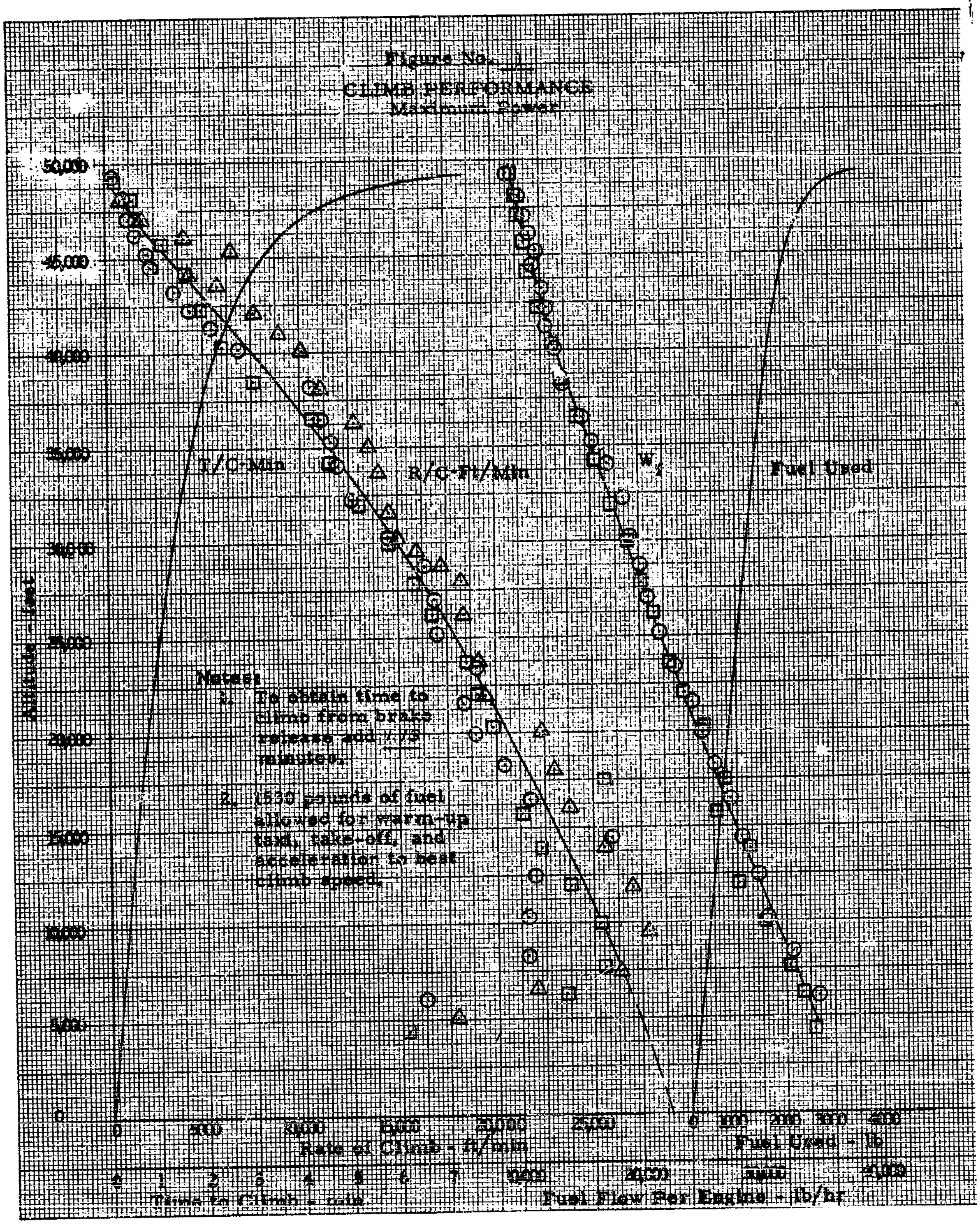
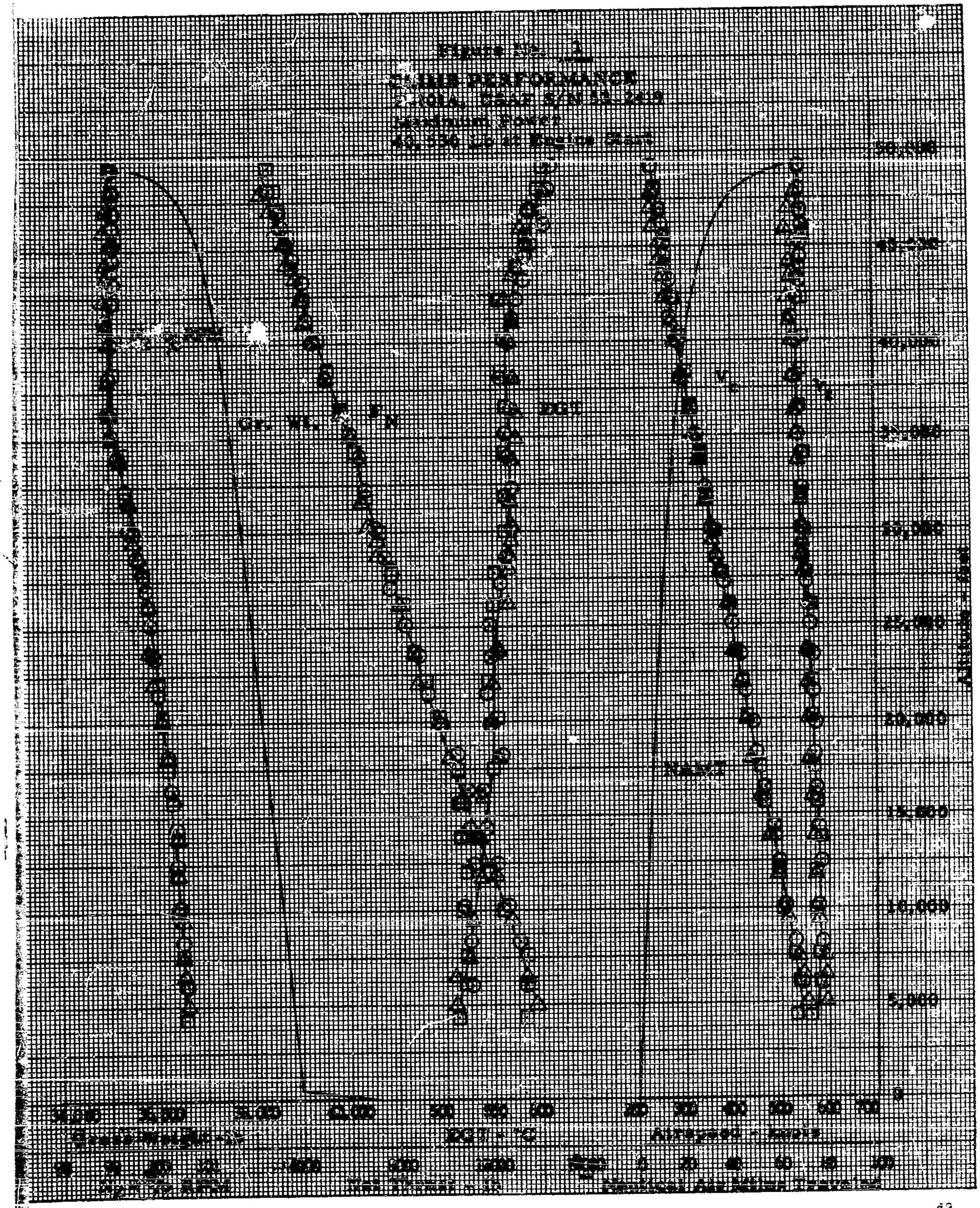


Figure No. 1
CLIMB PERFORMANCE
Maximum Power
Altitude - feet
50,000
45,000
40,000
35,000
30,000
25,000
20,000
15,000
10,000
5,000
0
T/C-Min
R/C-Ft/Min
W_f
Fuel Used
Notes:
1. To obtain time to climb from brake release add 1.13 minutes.
2. 1530 pounds of fuel allowed for warm-up taxi, take-off, and acceleration to best climb speed.
0 5000 10,000 15,000 20,000 25,000
Rate of Climb - ft/min
0 1000 2000 3000 4000
Fuel Used - lb
0 1 2 3 4 5 6 7
Time to Climb - min
10,000 20,000 30,000 40,000
Fuel Flow Per Engine - lb/hr


PERFORMANCE
Gr. Wt.
EGT
0
5,000
10,000
15,000
20,000
25,000
EGT - °C

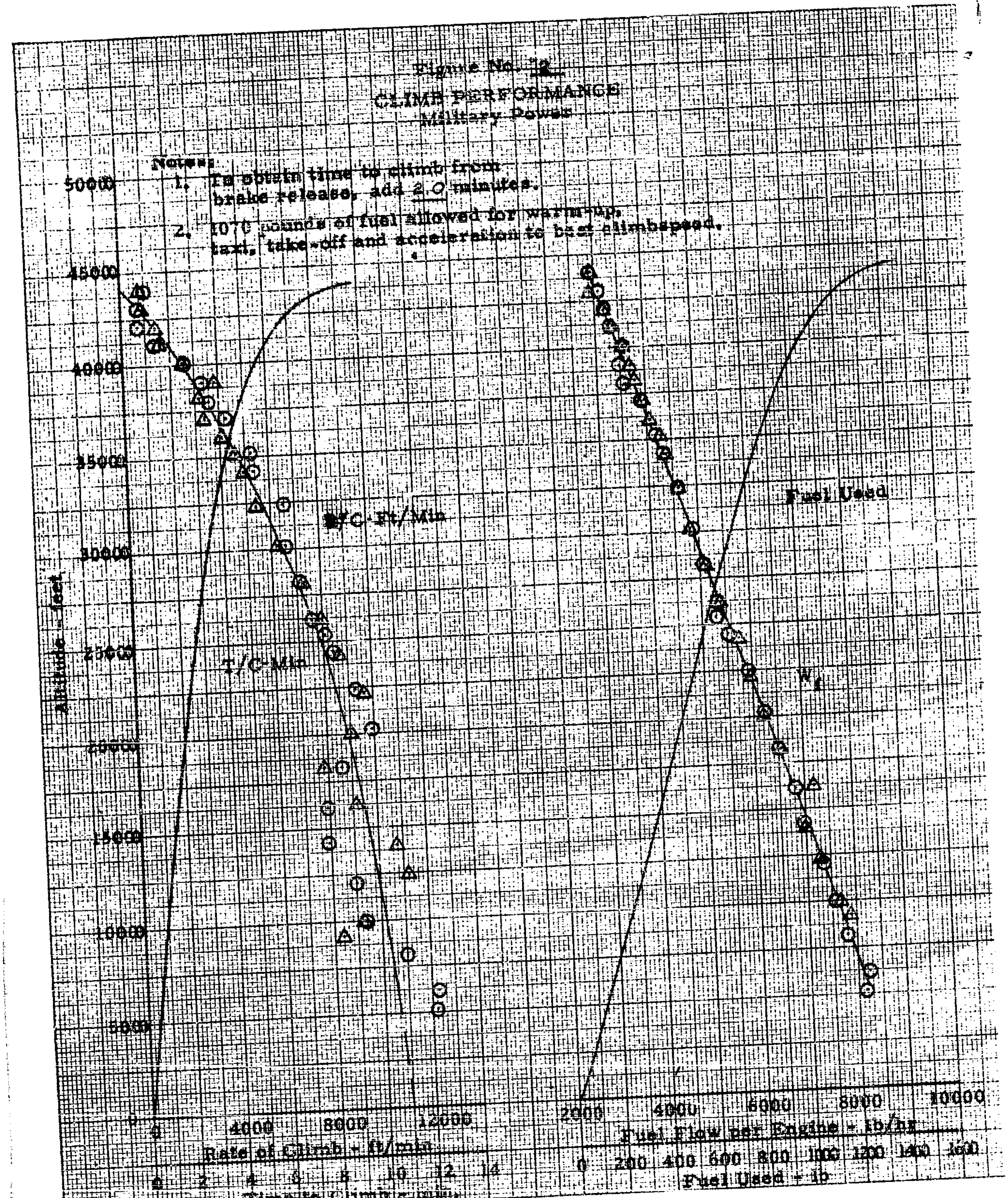


Figure No. 12
CLIMB PERFORMANCE
Military Power
Notes:
1. To obtain time to climb from brake release, add 2.0 minutes.
2. 1070 pounds of fuel allowed for warm-up, taxi, take-off and acceleration to best climbspeed.
R/C Ft/Min
T/C Min
Fuel Used
W_f
Altitude - feet
0
5000
10000
15000
20000
25000
30000
35000
40000
45000
50000
0
4000
8000
12000
Rate of Climb - ft/min
0
2
4
6
8
10
12
14
Time to Climb - min
2000
4000
6000
8000
10000
Fuel Flow per Engine - lb/hr
0
200
400
600
800
1000
1200
1400
1600
Fuel Used - lb

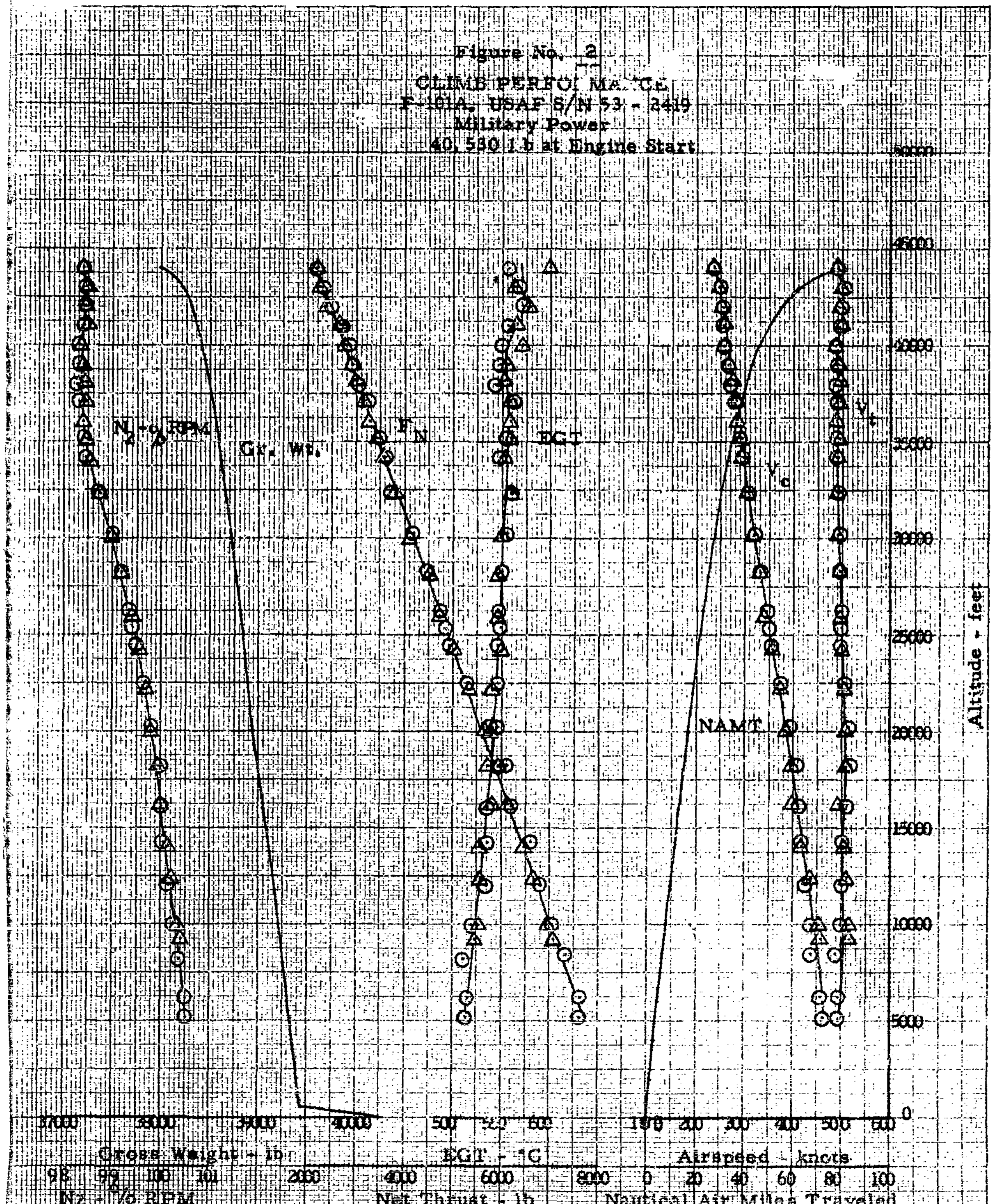
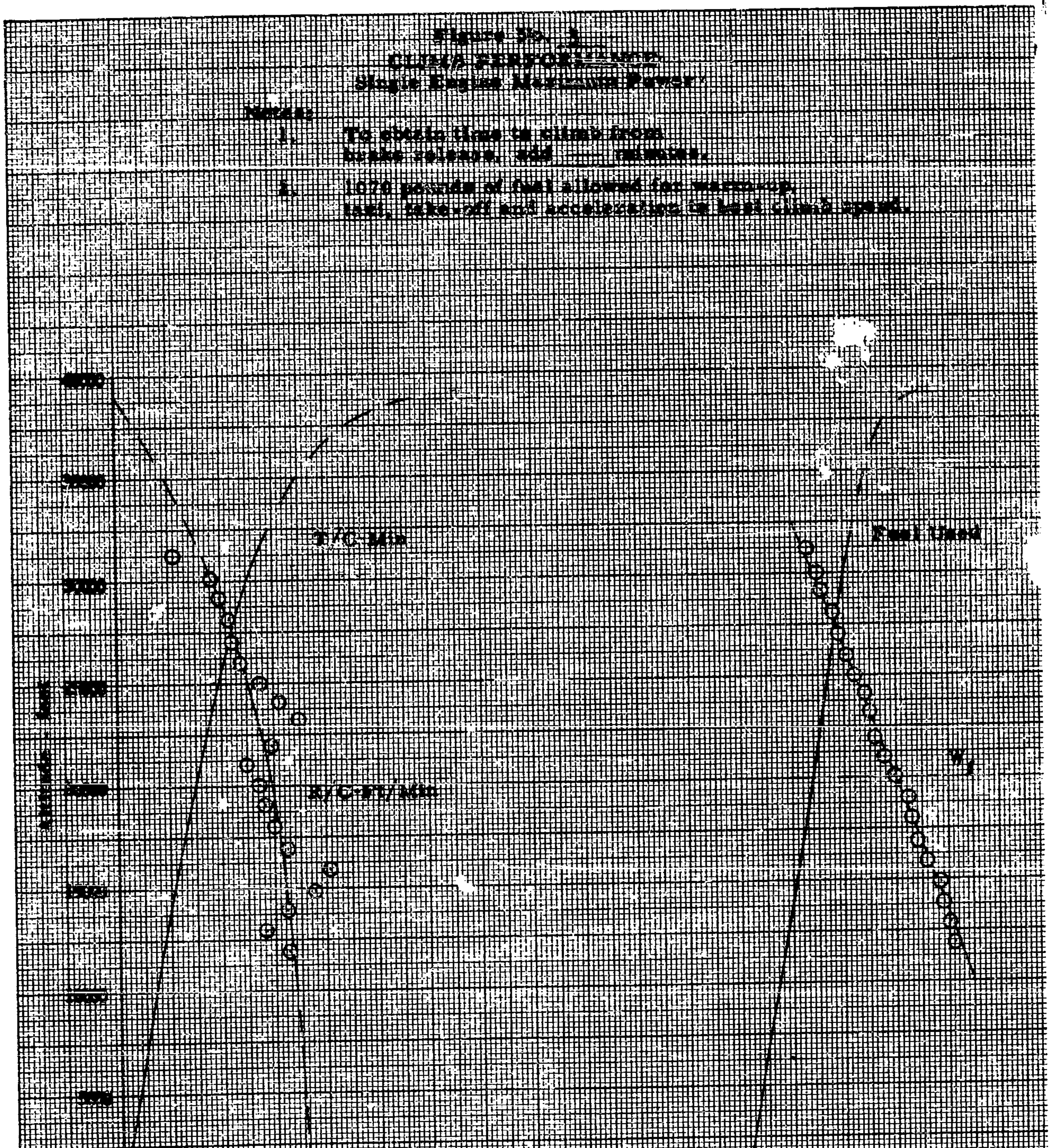
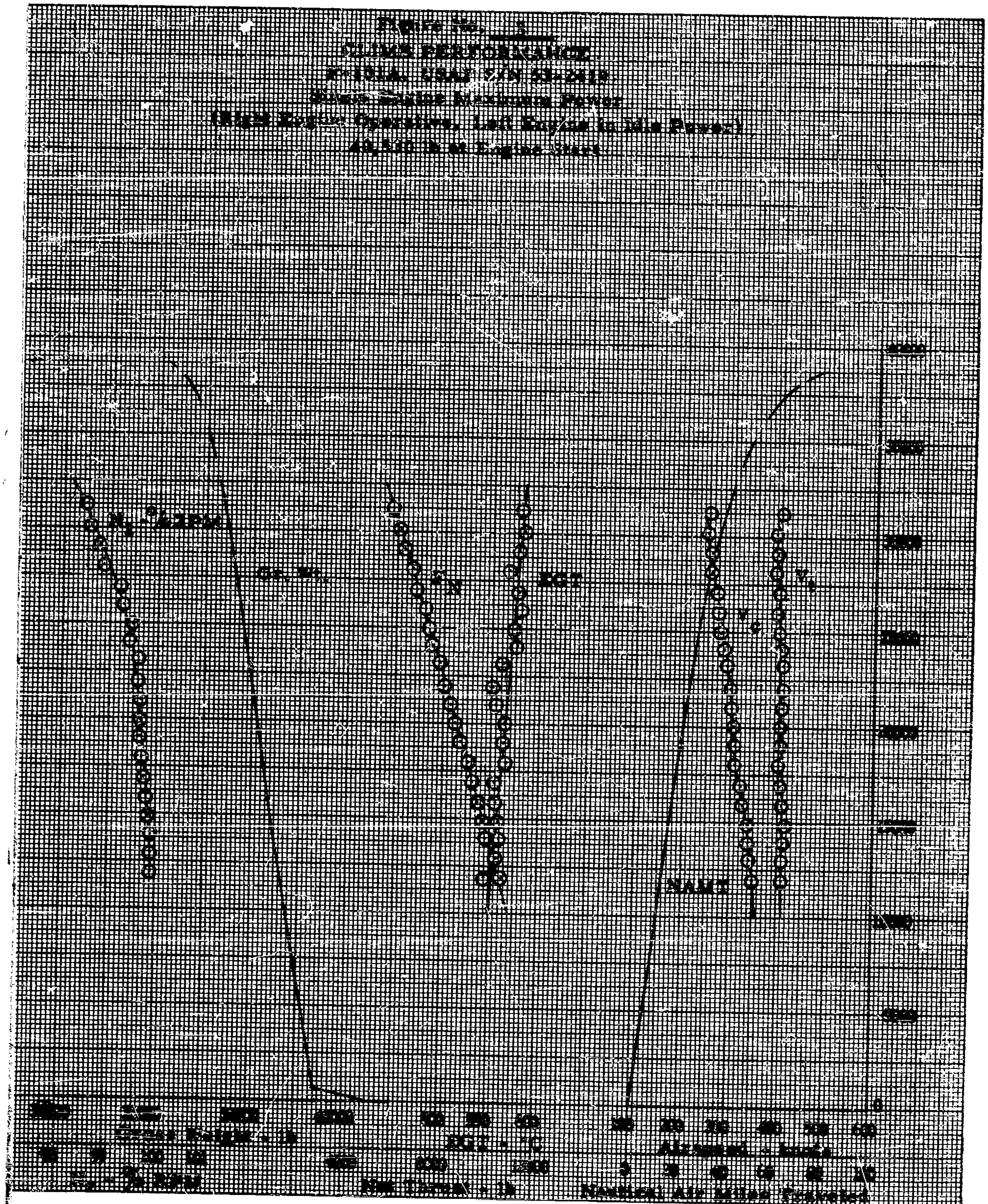


Figure No. 2
CLIMB PERFORMANCE
F-101A, USAF S/N 53 - 2419
Military Power
40,530 lb at Engine Start
N2 - % RPM
Gr. Wt.
FN
EGT
Vt
Vc
NAMT
Altitude - feet
0
5000
10000
15000
20000
25000
30000
35000
40000
45000
50000
37000
38000
39000
40000
500
520
600
100
200
300
400
500
600
Gross Weight - lb
EGT - °C
Airspeed - knots
98
99
100
101
2000
4000
6000
8000
0
20
40
60
80
100
N2 - % RPM
Net Thrust - lb
Nautical Air Miles Traveled


CLIMB PERFORMANCE
Single Engine Maximum Power
Notes:
1. To obtain time to climb from brake release, add ___ minutes.
2. 1070 pounds of fuel allowed for warm-up, taxi, take-off and acceleration to best climb speed.
T/C-Min
R/C-Ft/Min
Fuel Used
Altitude - feet
Rate of Climb - ft/min
Time to Climb - min.
Fuel Flow - lb/hr
Fuel Used - lb


CLIMB PERFORMANCE
Maximum Power
EGT
NAMT
0
EGT - °C
Nautical Air Miles Traveled

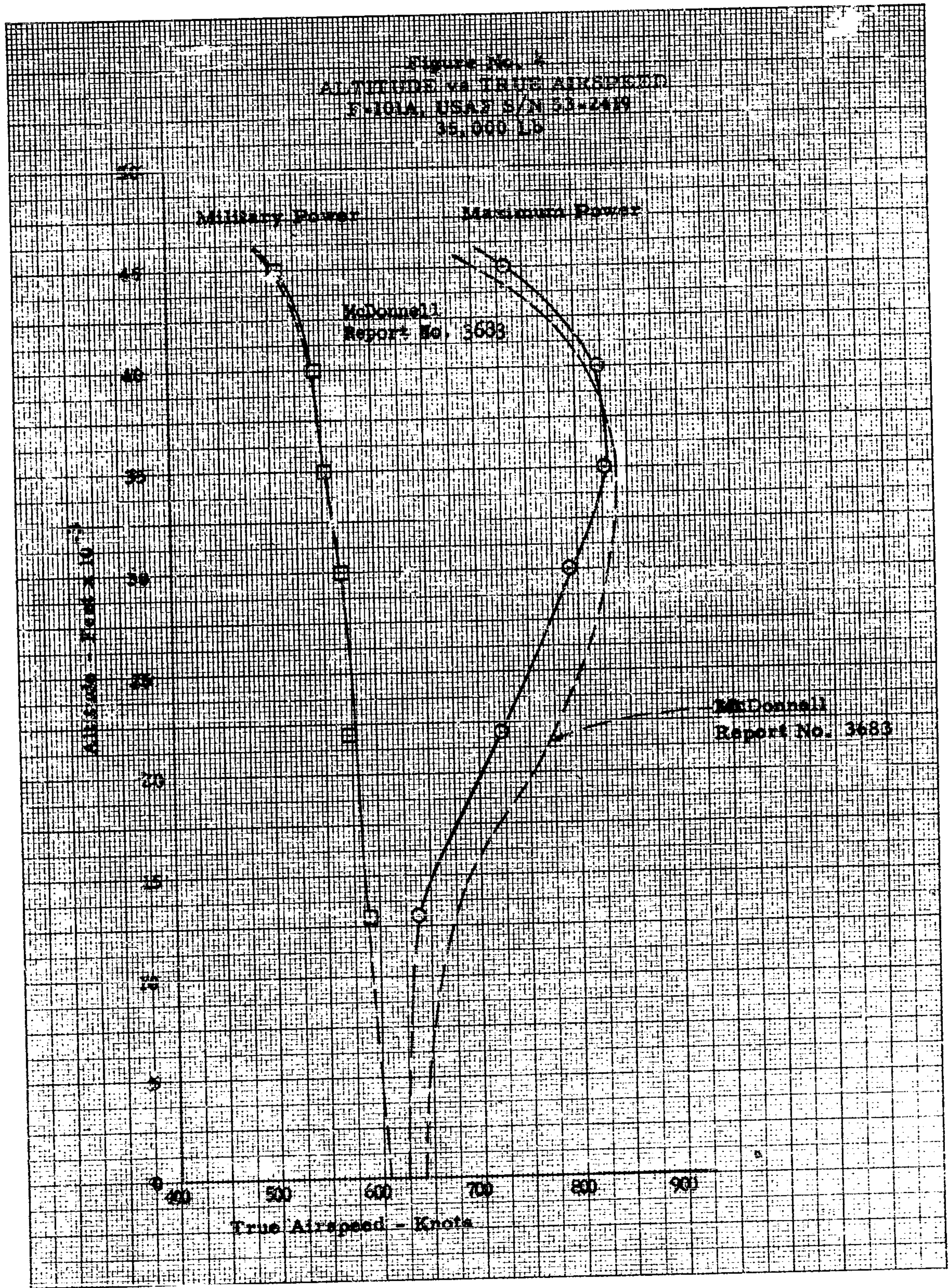


Figure No. 4
ALTITUDE vs TRUE AIRSPEED
F-101A, USAF S/N 53-2419
35,000 Lb
Military Power
Maximum Power
McDonnell
Report No. 3683
McDonnell
Report No. 3683
Altitude - Feet x 10^-3
50
45
40
35
30
25
20
15
10
5
0
400
500
600
700
800
900
True Airspeed - Knots

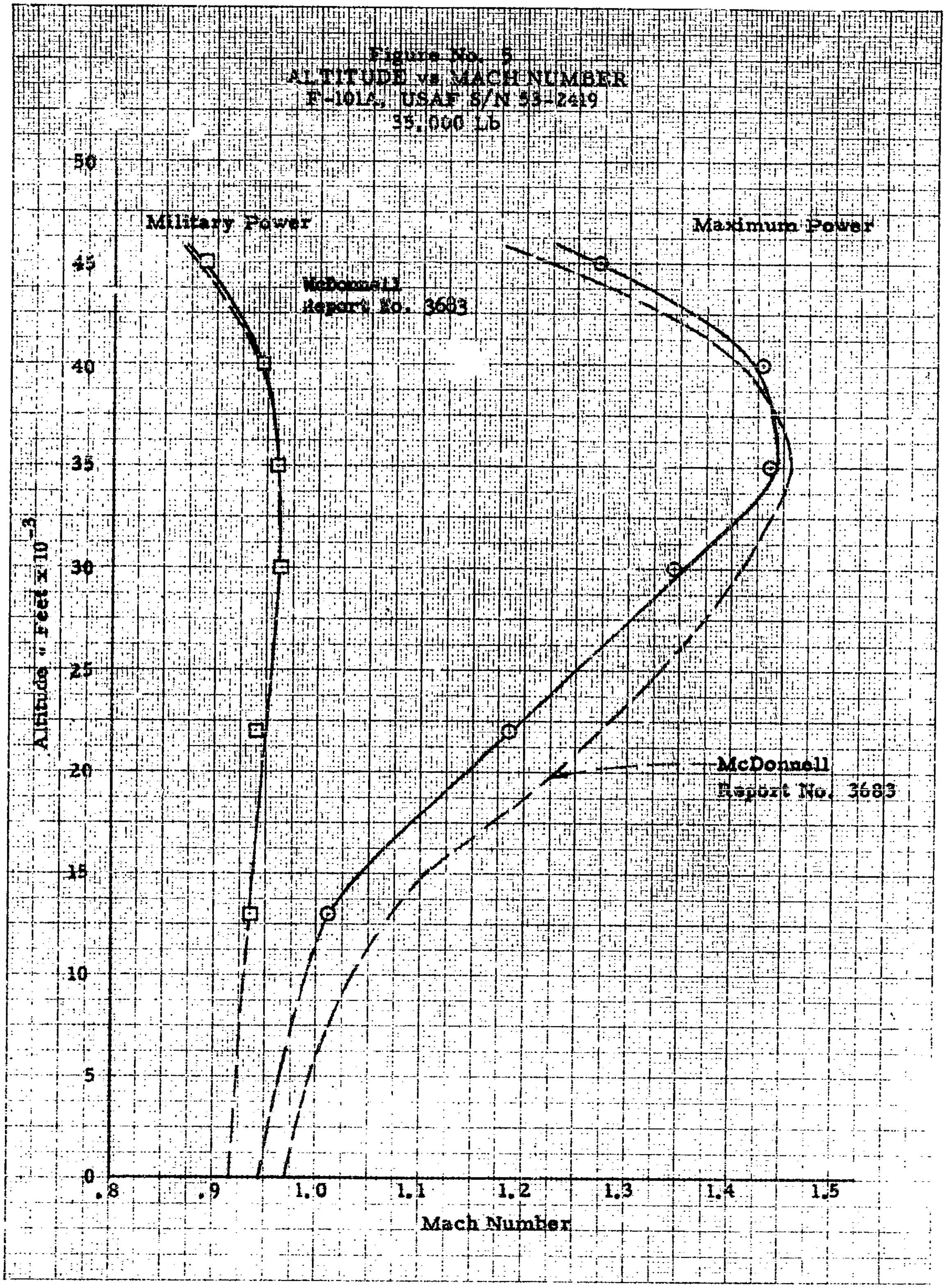
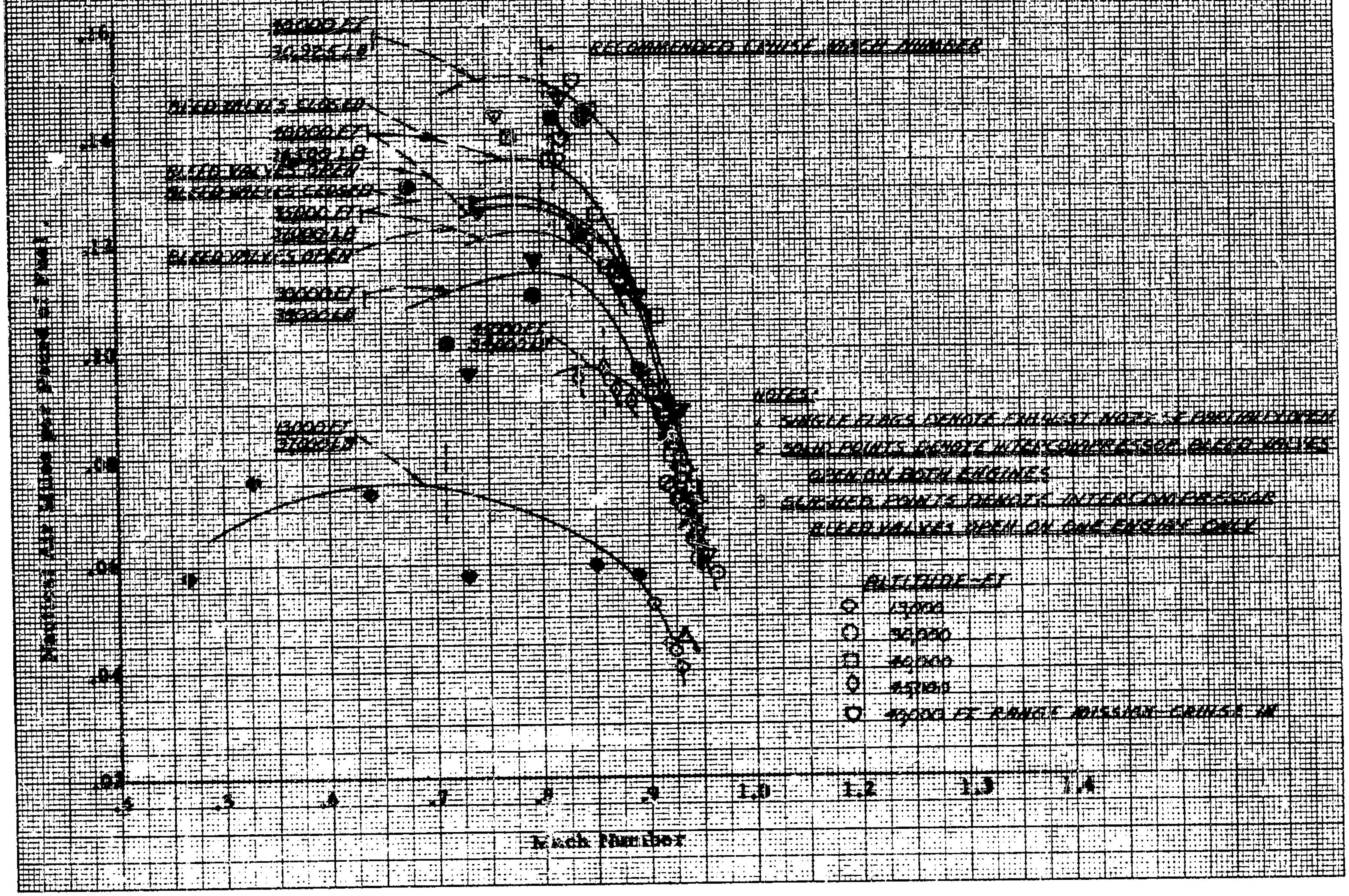


Figure No. 5
ALTITUDE vs MACH NUMBER
F-101A, USAF S/N 53-2419
35,000 Lb
Military Power
McDonnell
Report No. 3683
Maximum Power
McDonnell
Report No. 3683
Altitude - Feet x 10^-3
0
5
10
15
20
25
30
35
40
45
50
.8
.9
1.0
1.1
1.2
1.3
1.4
1.5
Mach Number

Figure No. 6
SPECIFIC RANGE vs. MACH NUMBER
F-101A, USAF S/N 53-2419



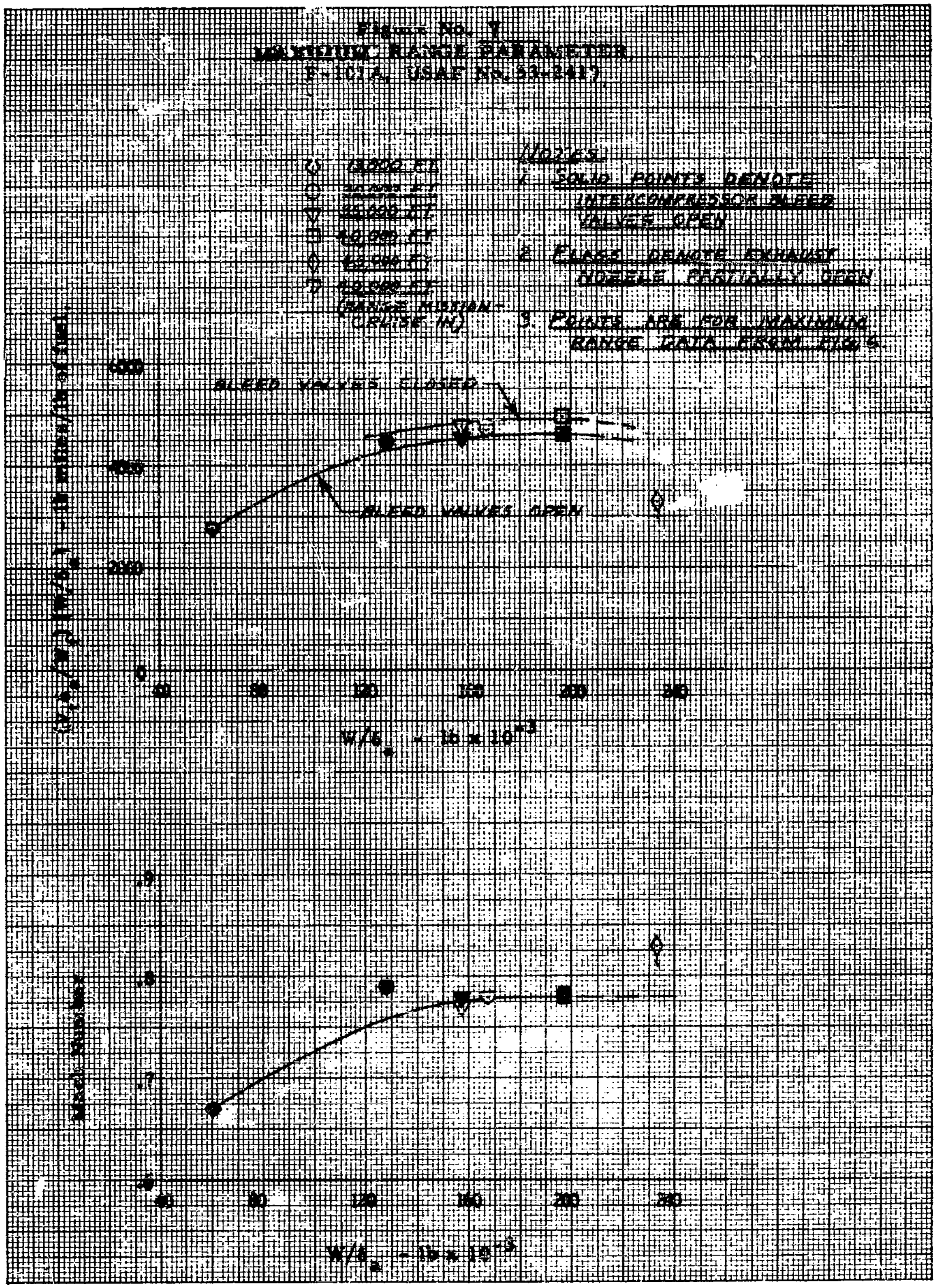


Figure No. 7
MAXIMUM RANGE PARAMETER
F-101A, USAF No. 53-2417
40,000 FT
(RANGE MISSION - CRUISE IN)
NOTES:
1. SOLID POINTS DENOTE INTERCOMPRESSOR BLEED VALVES OPEN
2. FLAGS DENOTE EXHAUST NOZZLE PARTIALLY OPEN
3. POINTS ARE FOR MAXIMUM RANGE DATA FROM FIG. 6
BLEED VALVES CLOSED
BLEED VALVES OPEN
6000
4000
2000
0
40
80
120
160
200
240
W/δa - lb x 10^-3
Mach Number
.9
.8
.7
.6

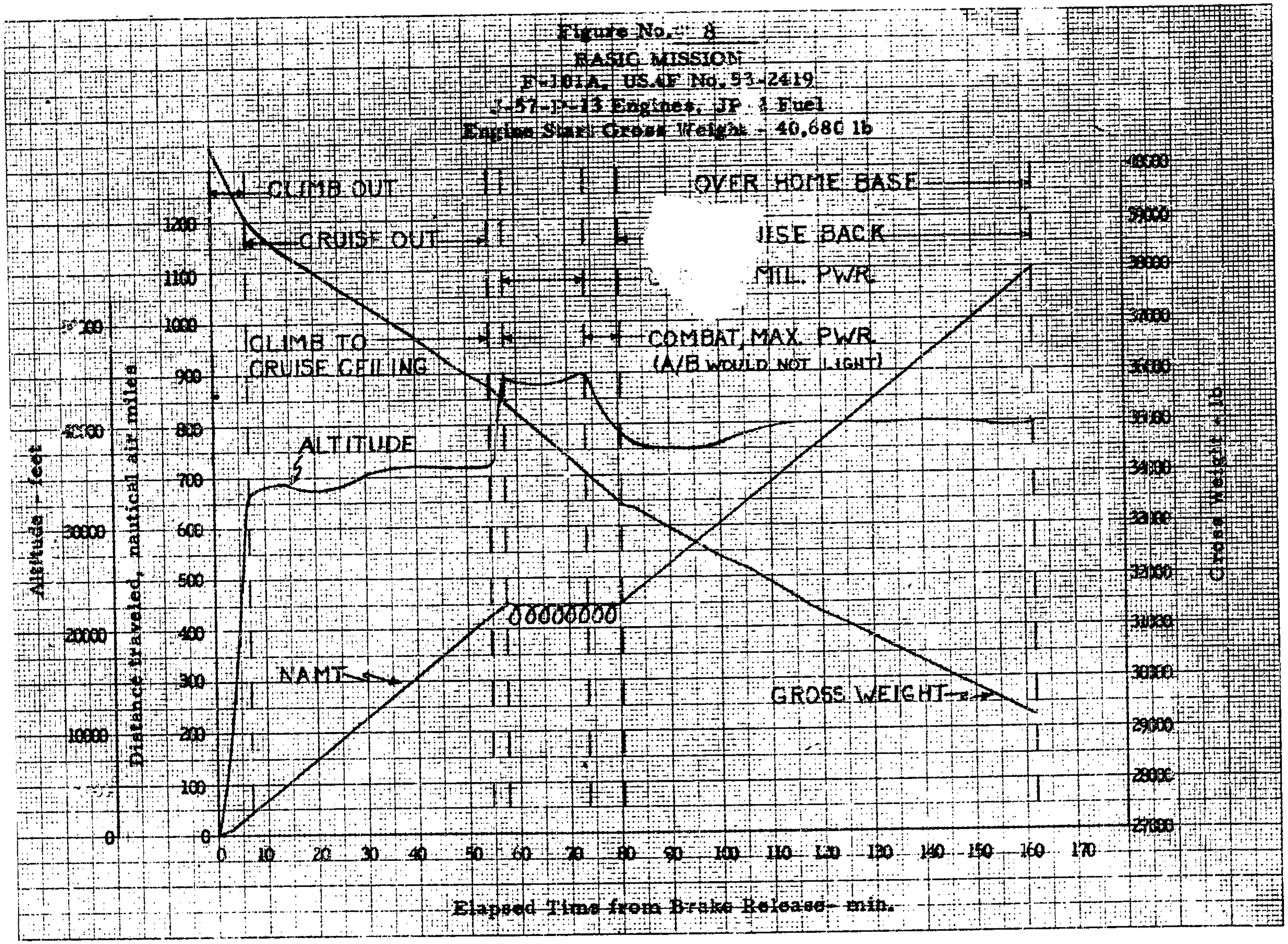


Figure No. 8
BASIC MISSION
F-101A, USAF No. 53-2419
J-57-P-13 Engines, JP 4 Fuel
Engine Start Gross Weight - 40,680 lb
CLIMB OUT
CRUISE OUT
CLIMB TO
CRUISE CEILING
OVER HOME BASE
MIL. PWR.
COMBAT, MAX. PWR.
(A/B WOULD NOT LIGHT)
ALTITUDE
NAMT
GROSS WEIGHT
Altitude - feet
Distance traveled, nautical air miles
Gross Weight - lb
Elapsed Time from Brake Release - min.

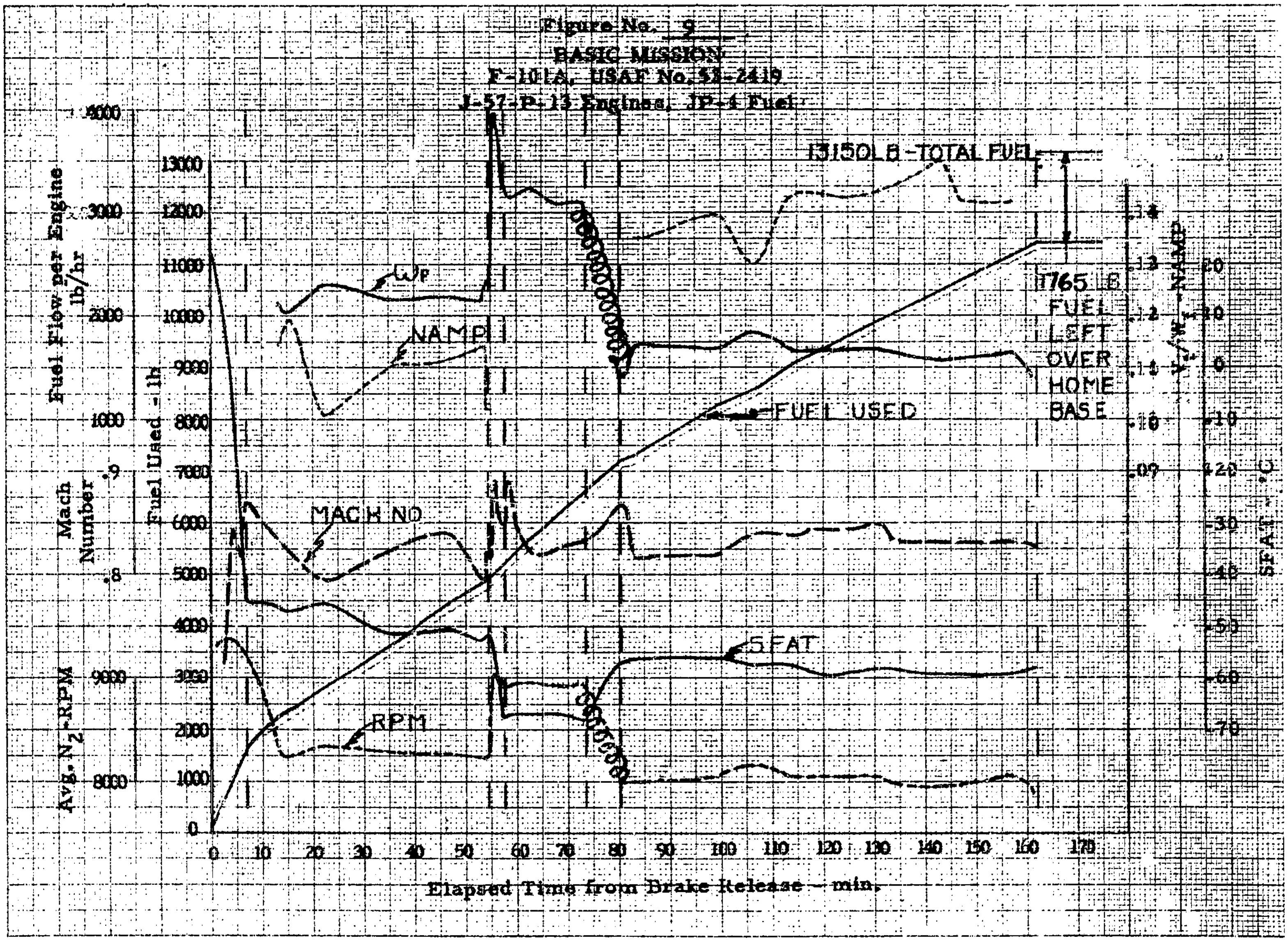
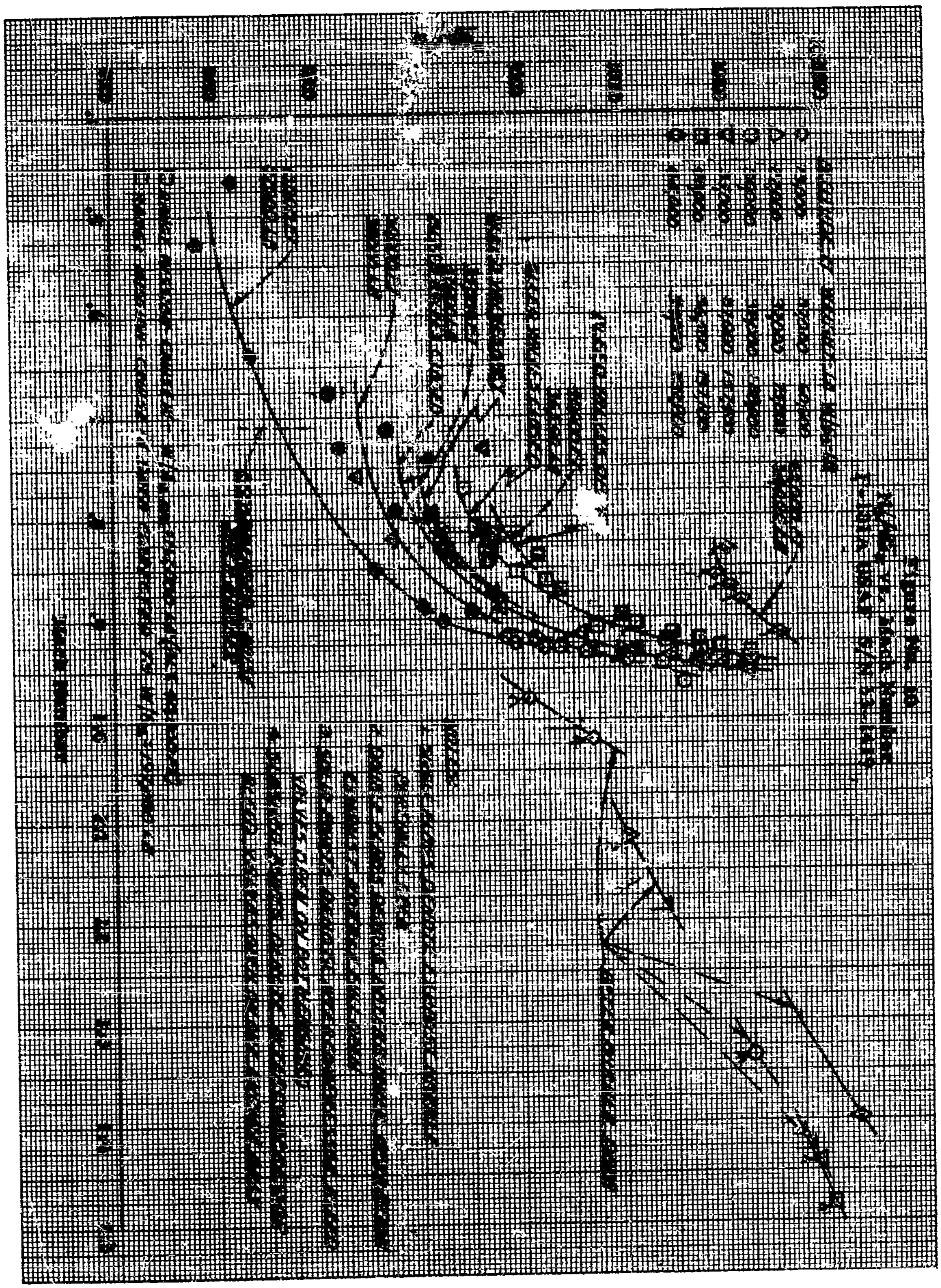


Figure No. 9
BASIC MISSION
F-101A, USAF No. 53-2419
J-57-P-13 Engines, JP-4 Fuel



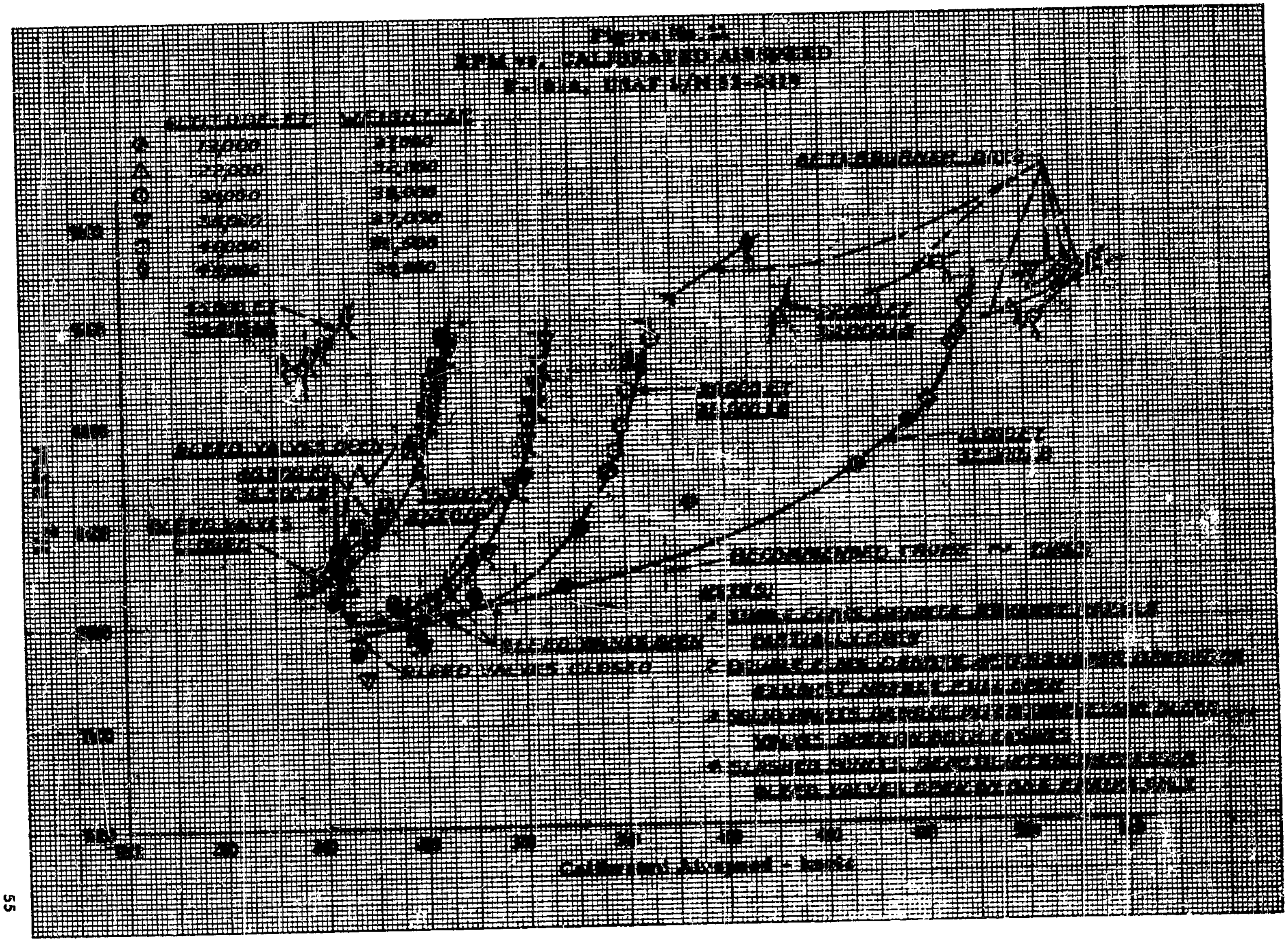
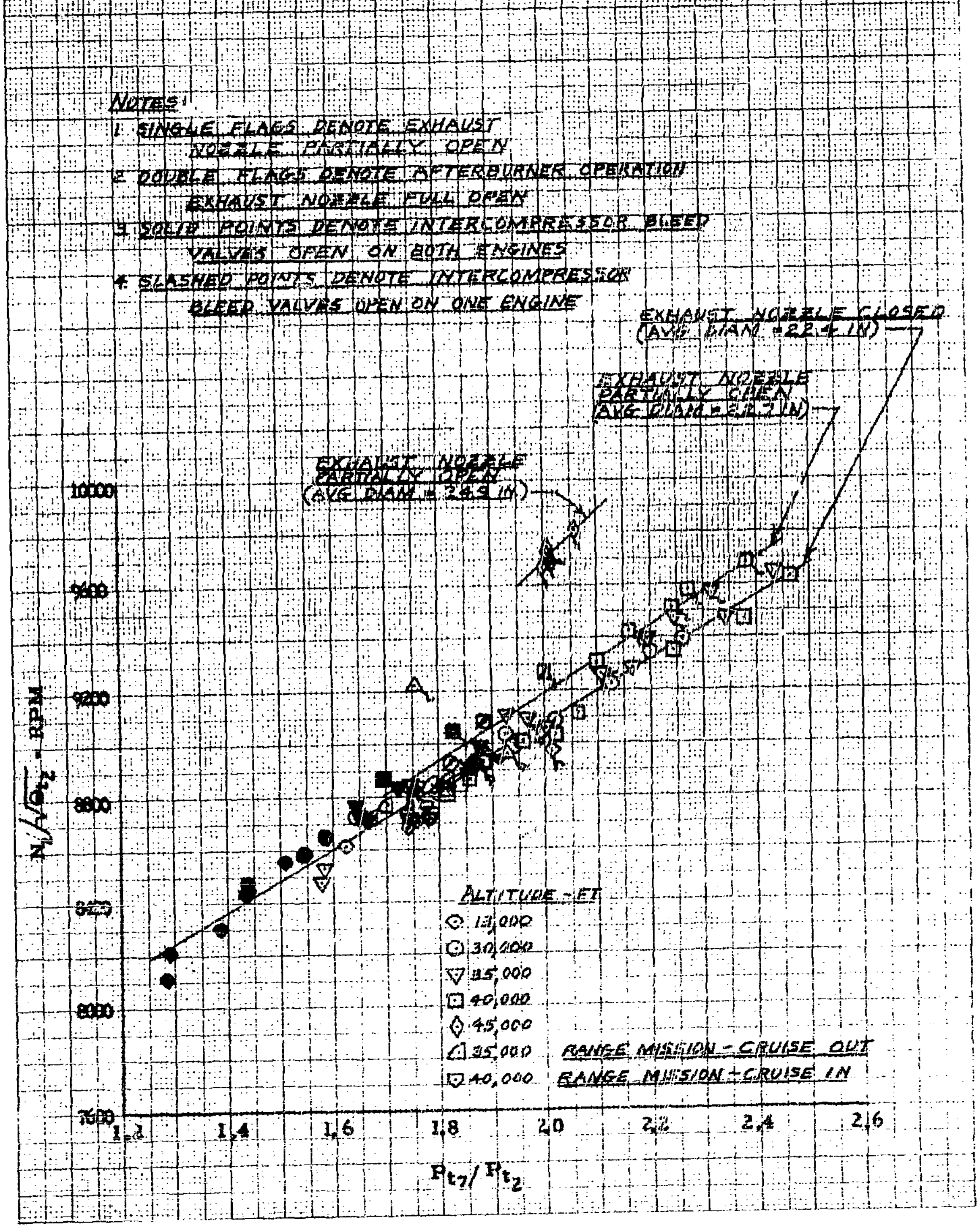

RPM vs. CALIBRATED AIRSPEED
NOTES:
Calibrated Airspeed - knots

Figure No. 12

$N_2/\sqrt{\theta_{t_2}}$ vs. P_{t_7}/P_{t_2}

F-101A, USAF S/N 53-2419



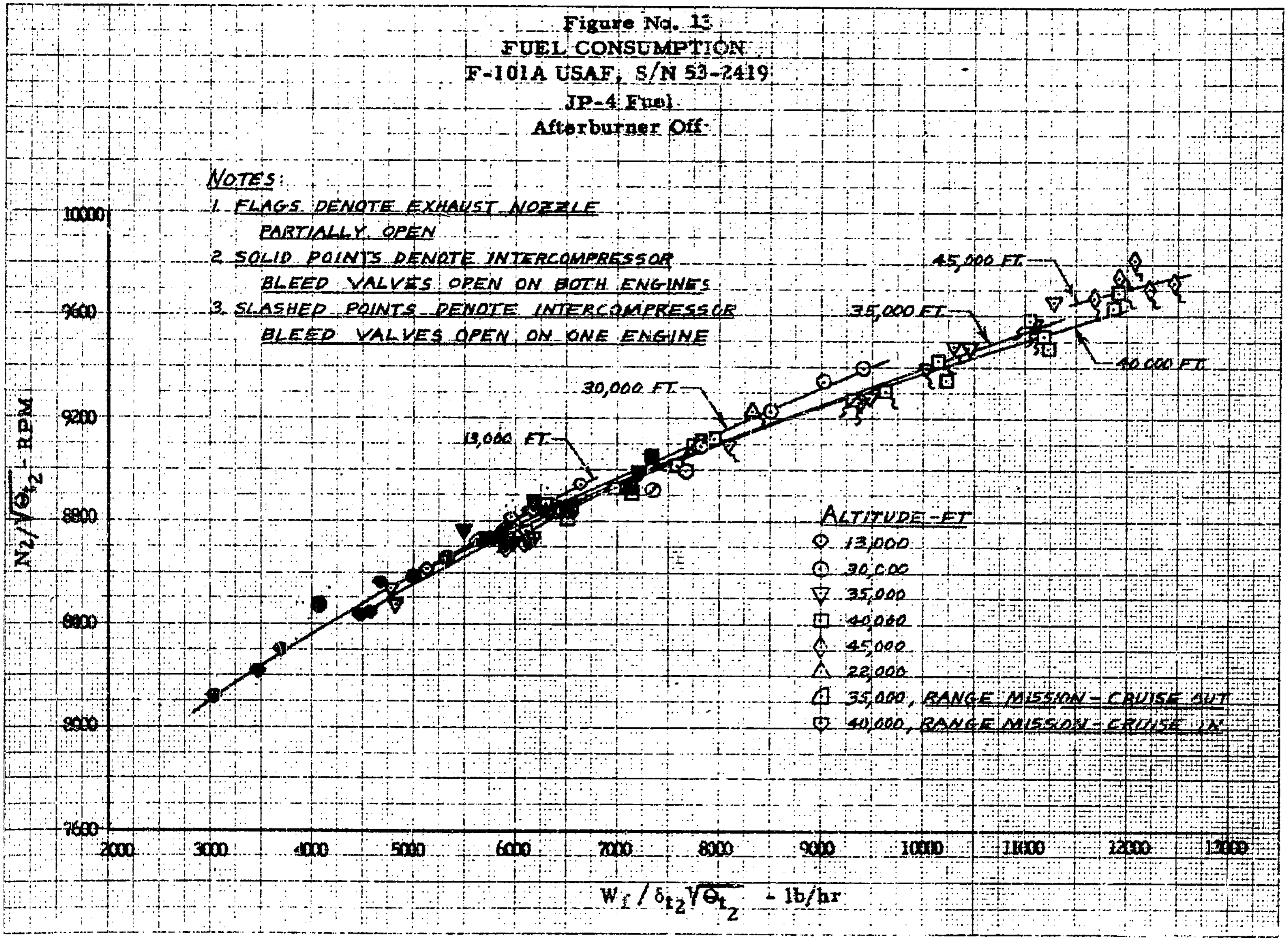
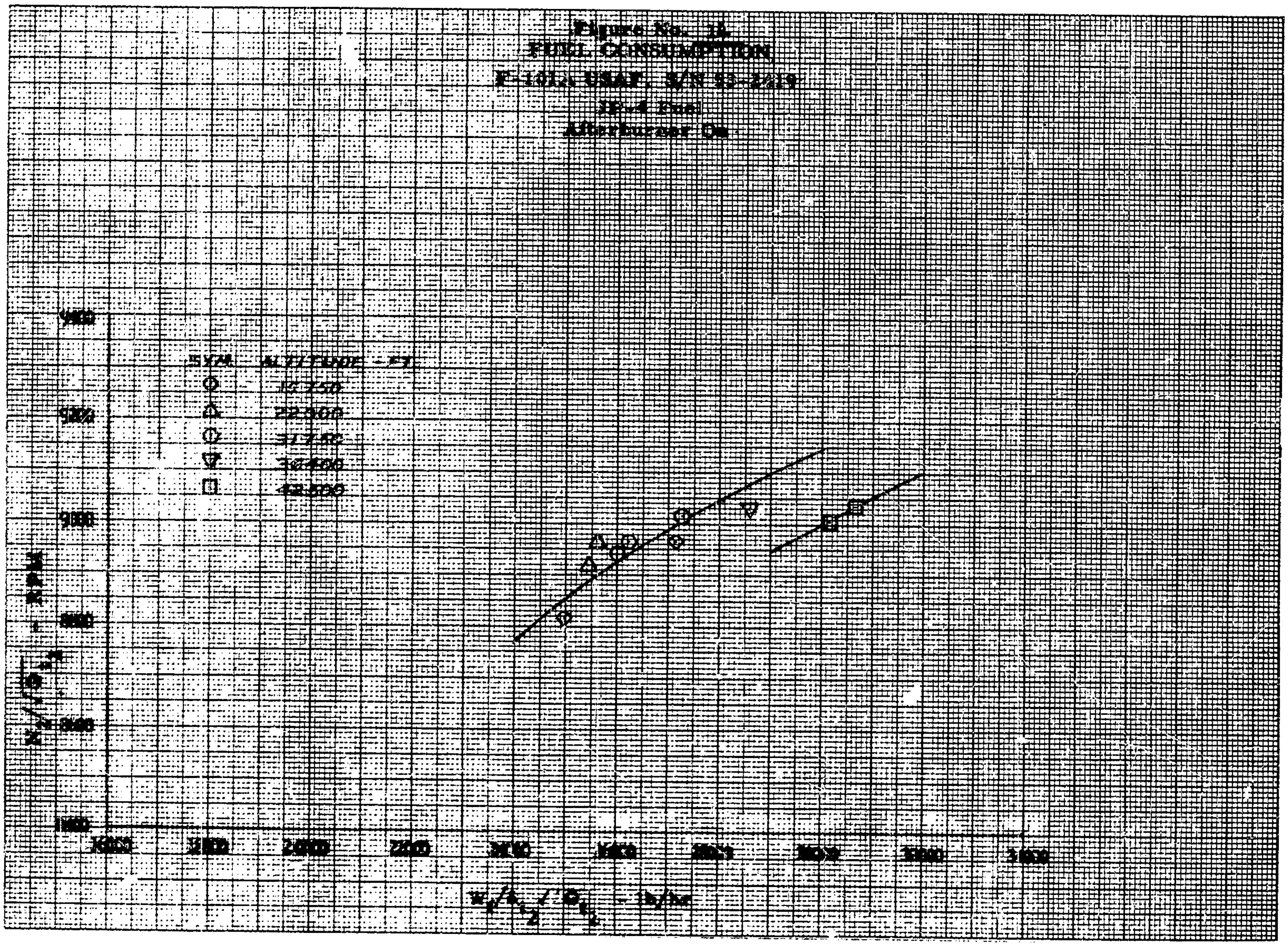


Figure No. 13
FUEL CONSUMPTION
F-101A USAF, S/N 53-2419
JP-4 Fuel
Afterburner Off


FUEL CONSUMPTION
JP-4 Fuel
Afterburner On
SYM ALTITUDE - FT
22500
36400
42500

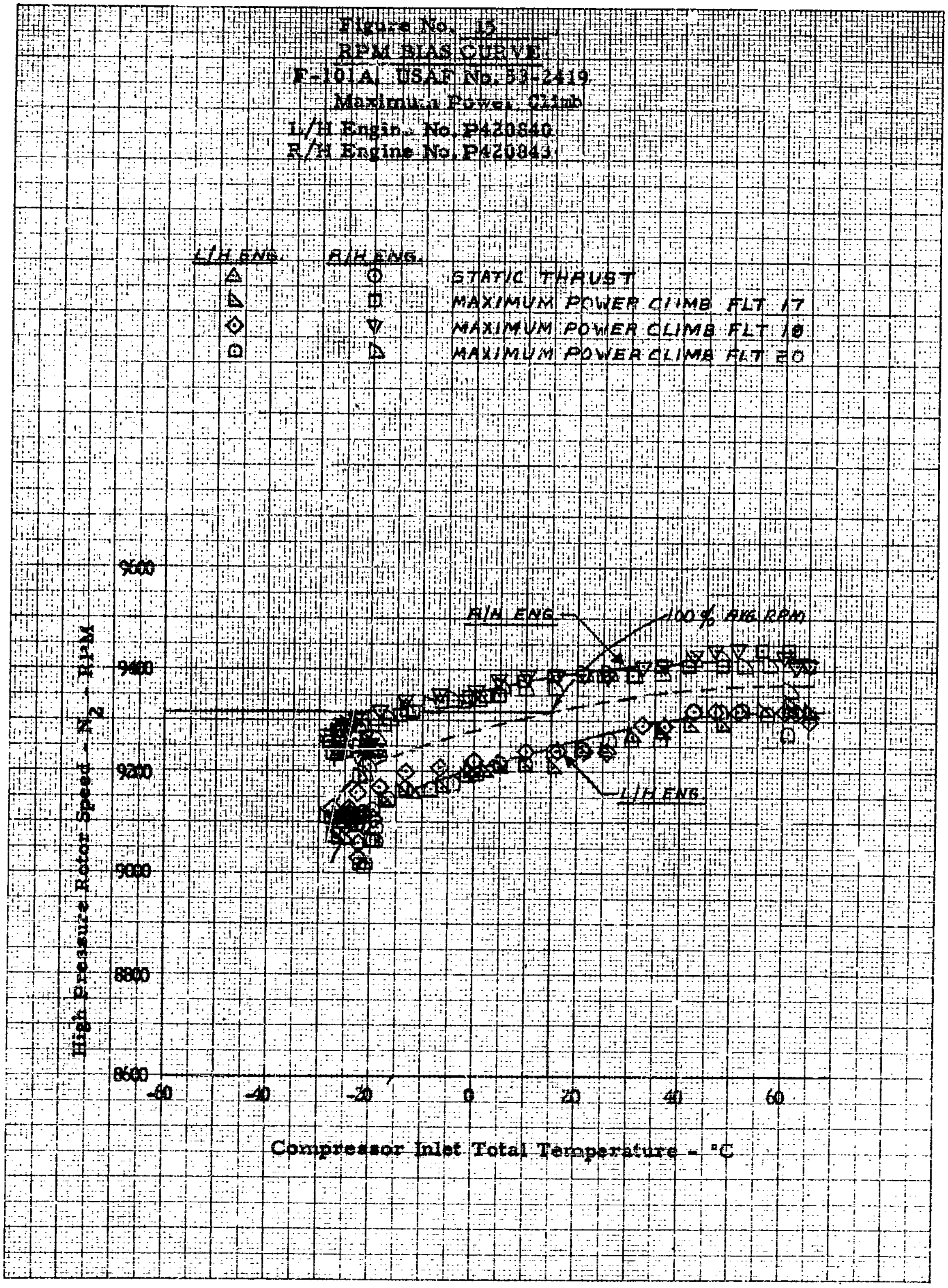


Figure No. 15
RPM BIAS CURVE
F-101A USAF No. 53-2419
Maximum Power Climb
L/H Engine No. P420840
R/H Engine No. P420843
L/H ENG.
R/H ENG.
STATIC THRUST
MAXIMUM POWER CLIMB FLT 17
MAXIMUM POWER CLIMB FLT 18
MAXIMUM POWER CLIMB FLT 20
R/H ENG.
100 % AVG RPM
L/H ENG.
High Pressure Rotor Speed - N_2 - RPM
9600
9400
9200
9000
8800
8600
-60
-40
-20
0
20
40
60
Compressor Inlet Total Temperature - °C

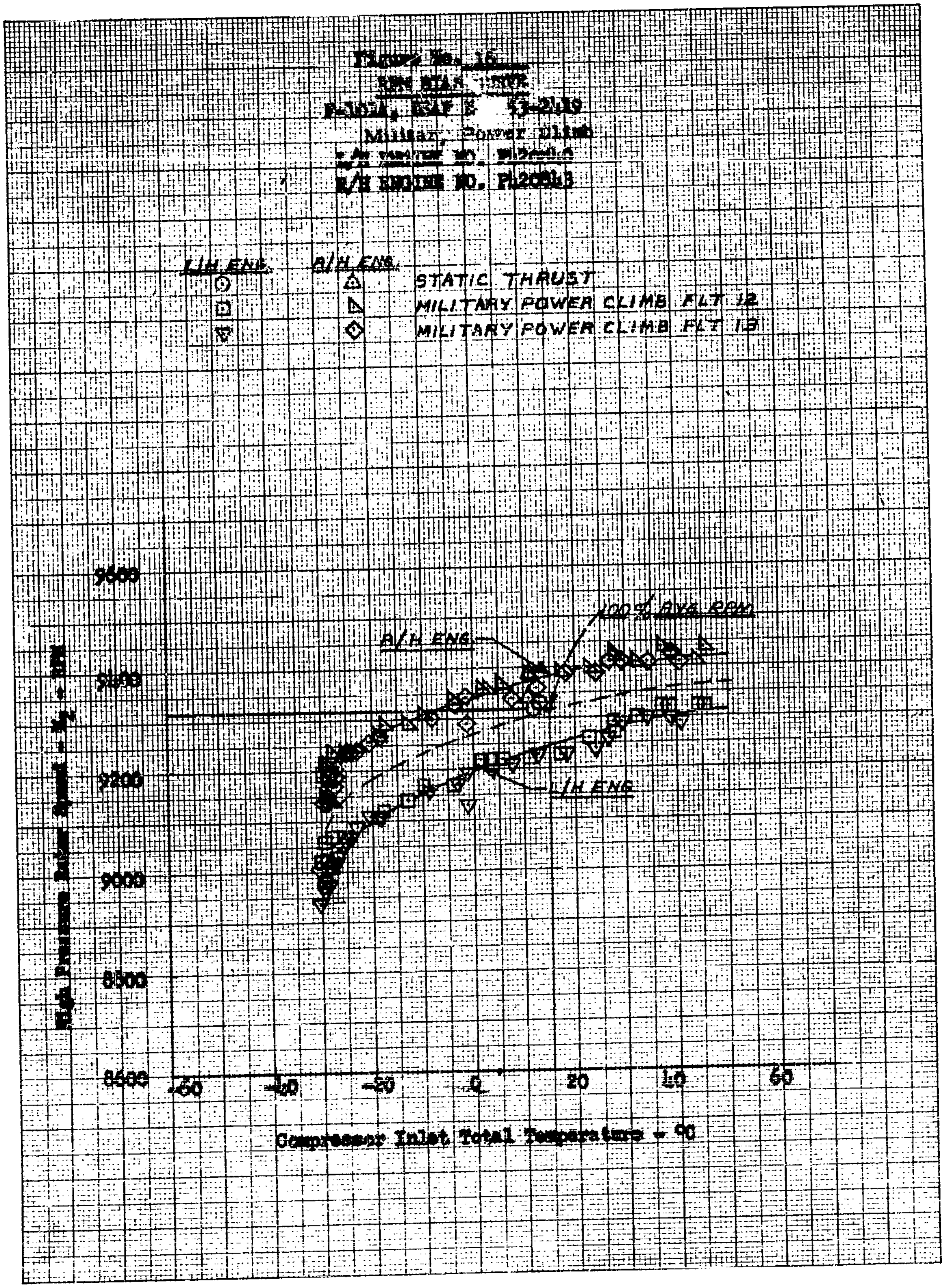


Figure No. 16
Military Power Climb
R/H ENGINE NO.
L/H ENG.
R/H ENG.
STATIC THRUST
MILITARY POWER CLIMB FLT 12
MILITARY POWER CLIMB FLT 13
100% AXE RPM
R/H ENG.
L/H ENG.
9600
9400
9200
9000
8800
8600
-60
-40
-20
0
20
40
60
Compressor Inlet Total Temperature - °C

Figure No. 17

RPM BIAS CURVE

F-101A, USAF No. 53-2419

Level Flight Maximum and Military Power,

L/H Engine No. P420840

R/H Engine No. P420843

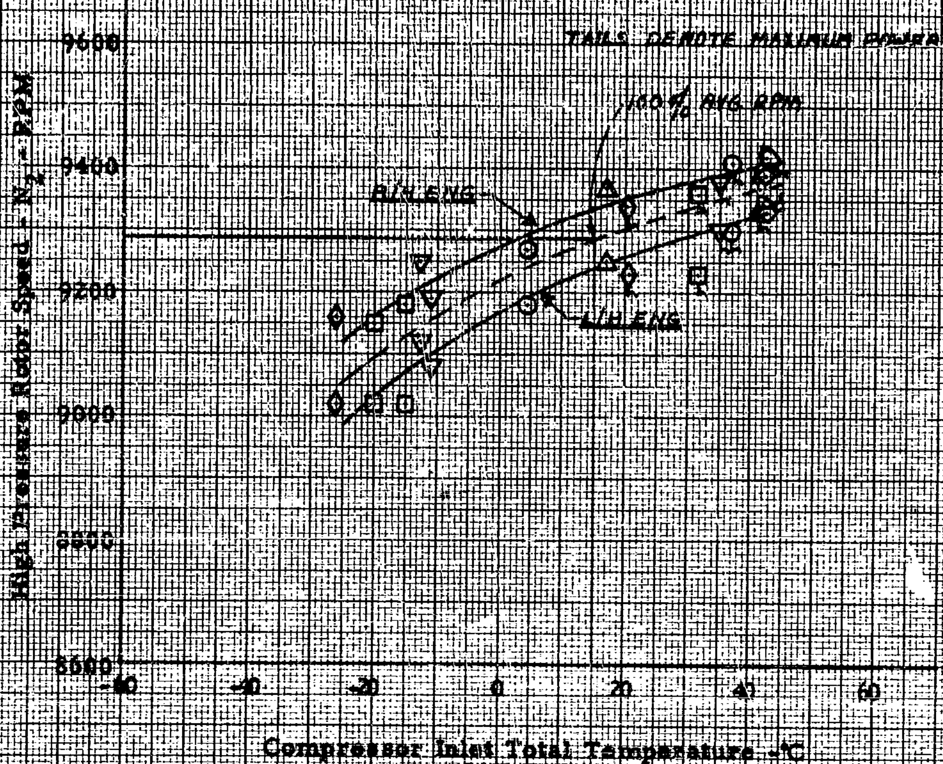
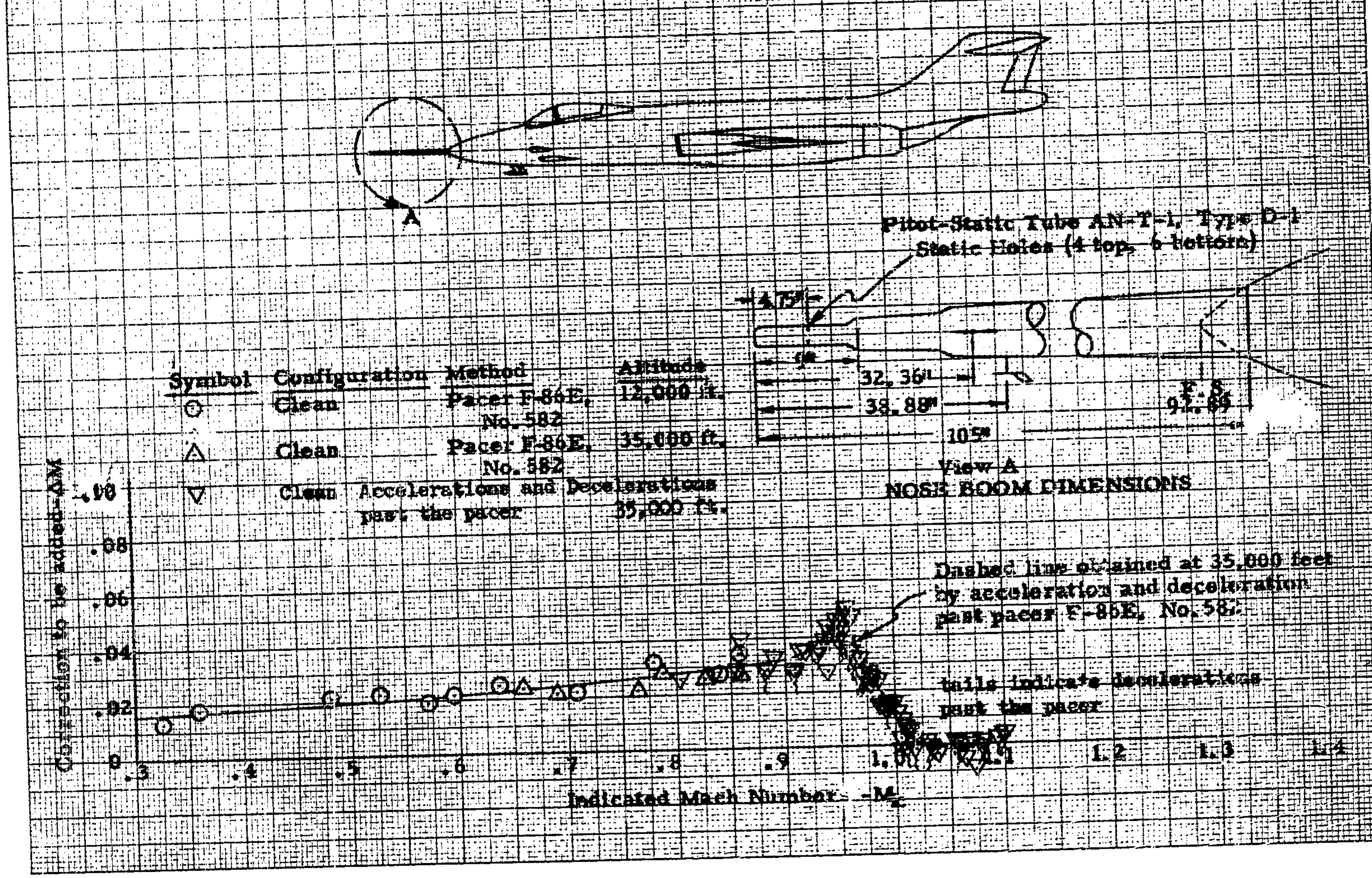


Figure No. 18
AIRSPEED CALIBRATION
Nose Boom System
F-101A, USAF No. 53-2419



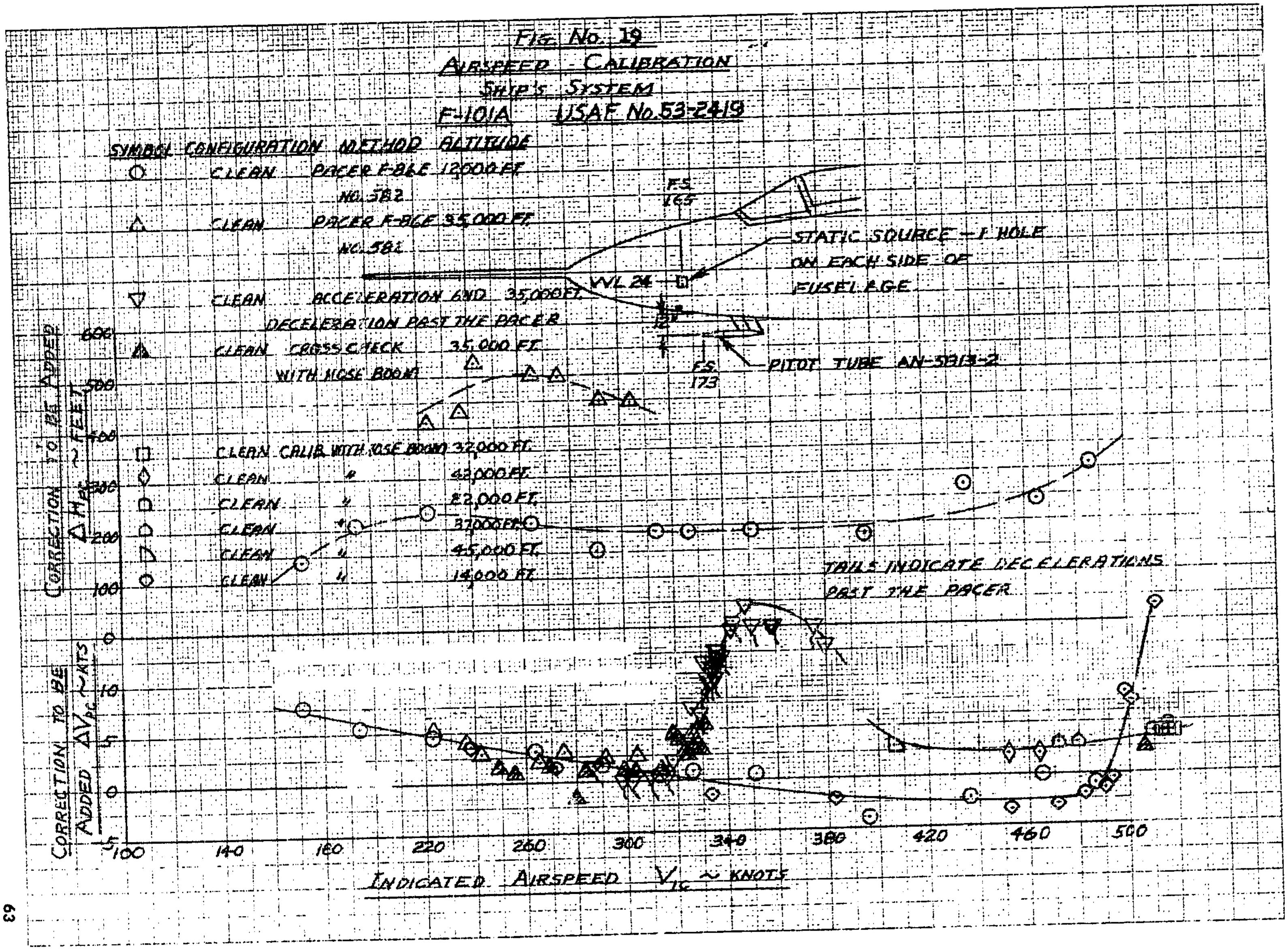


FIG. NO. 19
AIRSPEED - CALIBRATION
SHIP'S SYSTEM
F-101A USAF No. 53-2419
SYMBOL CONFIGURATION METHOD ALTITUDE
CLEAN PACER F-86E NO. 582 12,000 FT.
CLEAN PACER F-86E NO. 582 35,000 FT.
CLEAN ACCELERATION AND DECELERATION PAST THE PACER 35,000 FT.
CLEAN CROSS CHECK WITH NOSE BOOM 35,000 FT.
CLEAN CALIB. WITH NOSE BOOM 32,000 FT.
CLEAN " 42,000 FT.
CLEAN " 22,000 FT.
CLEAN " 37,000 FT.
CLEAN " 45,000 FT.
CLEAN " 14,000 FT.
F.S. 165
F.S. 173
WL 24
STATIC SOURCE - 1 HOLE ON EACH SIDE OF FUSELAGE
PITOT TUBE AN-5813-2
TAILS INDICATE DECELERATIONS PAST THE PACER
CORRECTION TO BE ADDED ΔH_{pc} ~ FEET
CORRECTION TO BE ADDED ΔV_{pc} ~ KTS
INDICATED AIRSPEED V_{ic} ~ KNOTS
600
500
400
300
200
100
0
10
5
0
-5
100
140
180
220
260
300
340
380
420
460
500

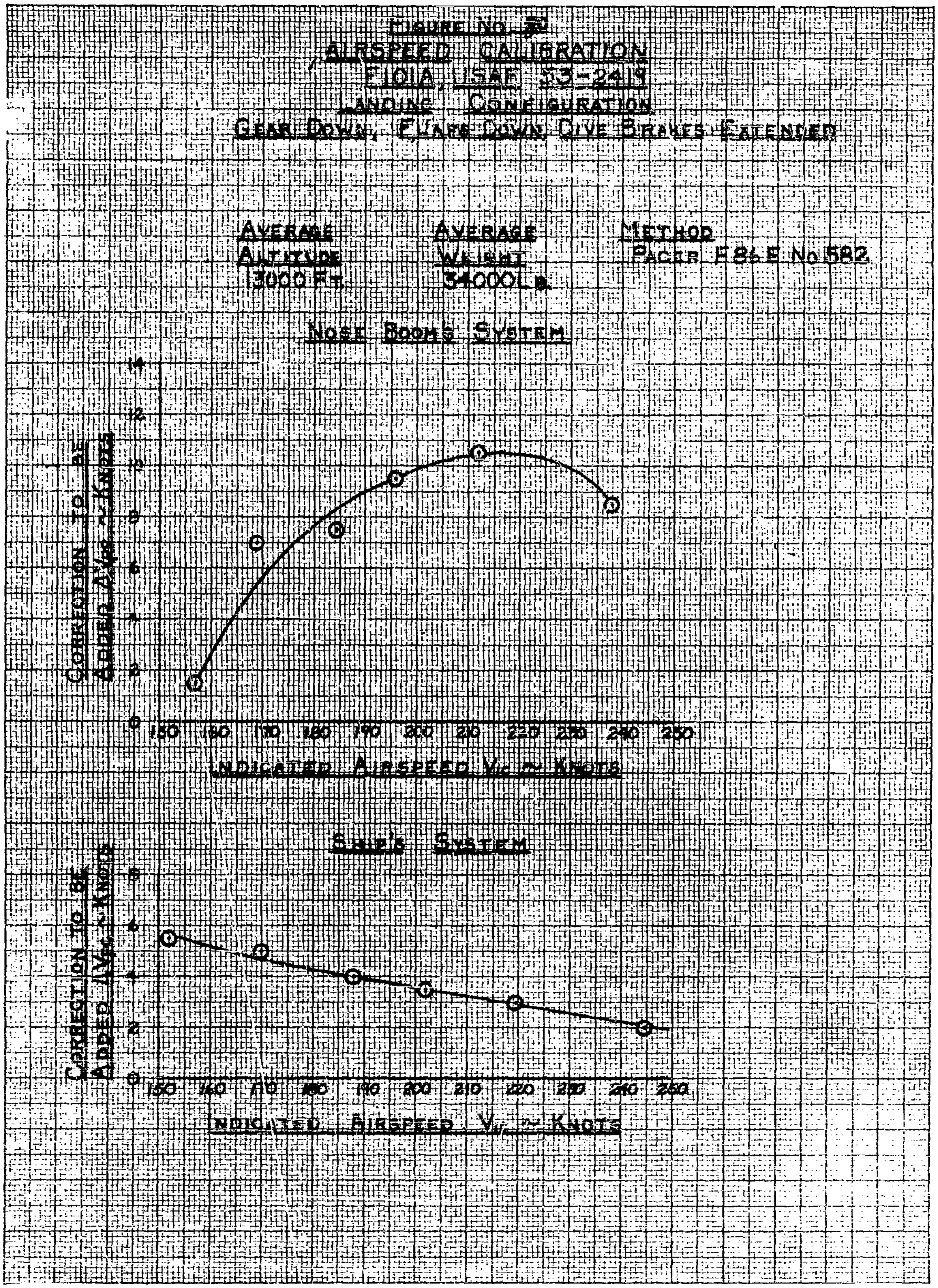


FIGURE NO.
AIRSPEED CALIBRATION
F101A, USAF 53-2419
LANDING CONFIGURATION
GEAR DOWN, FLAPS DOWN, DIVE BRAKES EXTENDED
AVERAGE ALTITUDE 13000 FT.
AVERAGE WEIGHT 34000 LB.
METHOD PACER F86E No 582
NOSE BOOM'S SYSTEM
CORRECTION TO BE ADDED ΔV_{pc} ~ KNOTS
0 2 4 6 8 10 12 14
150 160 170 180 190 200 210 220 230 240 250
INDICATED AIRSPEED V_{ic} ~ KNOTS
SHIP'S SYSTEM
CORRECTION TO BE ADDED ΔV_{pc} ~ KNOTS
0 2 4 6 8
150 160 170 180 190 200 210 220 230 240 250
INDICATED AIRSPEED V_{ic} ~ KNOTS

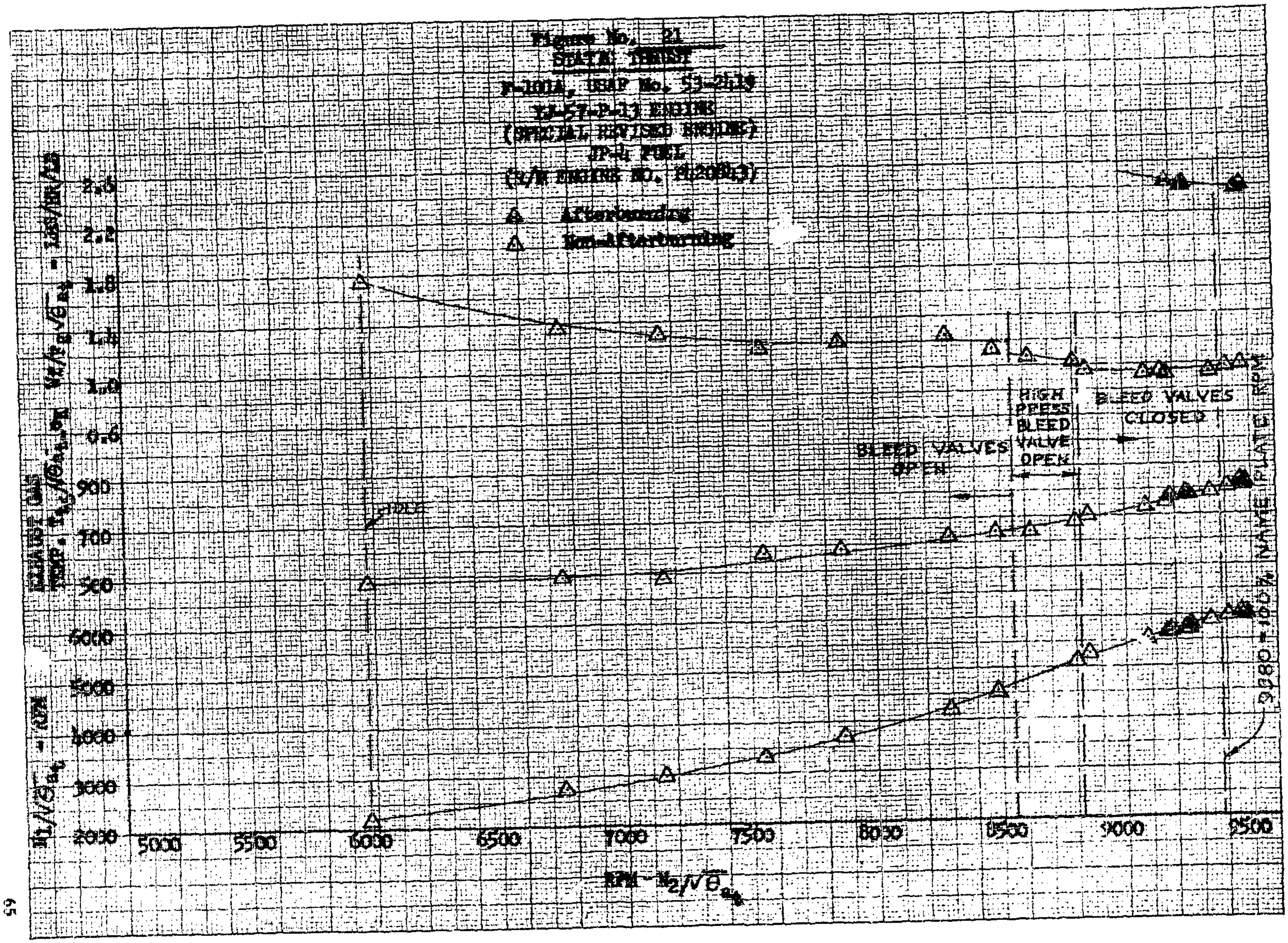


Figure No. 21
STATIC THRUST
F-101A, USAF No. 53-2419
YJ-57-P-13 ENGINE
(SPECIAL REVISED ENGINE)
JP-4 FUEL
Afterburning
Non-Afterburning
IDLE
BLEED VALVES OPEN
HIGH PRESS BLEED VALVE OPEN
BLEED VALVES CLOSED
9880 = 100% NAME PLATE RPM
2.6
2.2
1.8
1.4
1.0
0.6
900
700
500
6000
5000
4000
3000
2000
5000
5500
6000
6500
7000
7500
8000
8500
9000
9500
RPM - $N_2/\sqrt{\theta_{a_4}}$

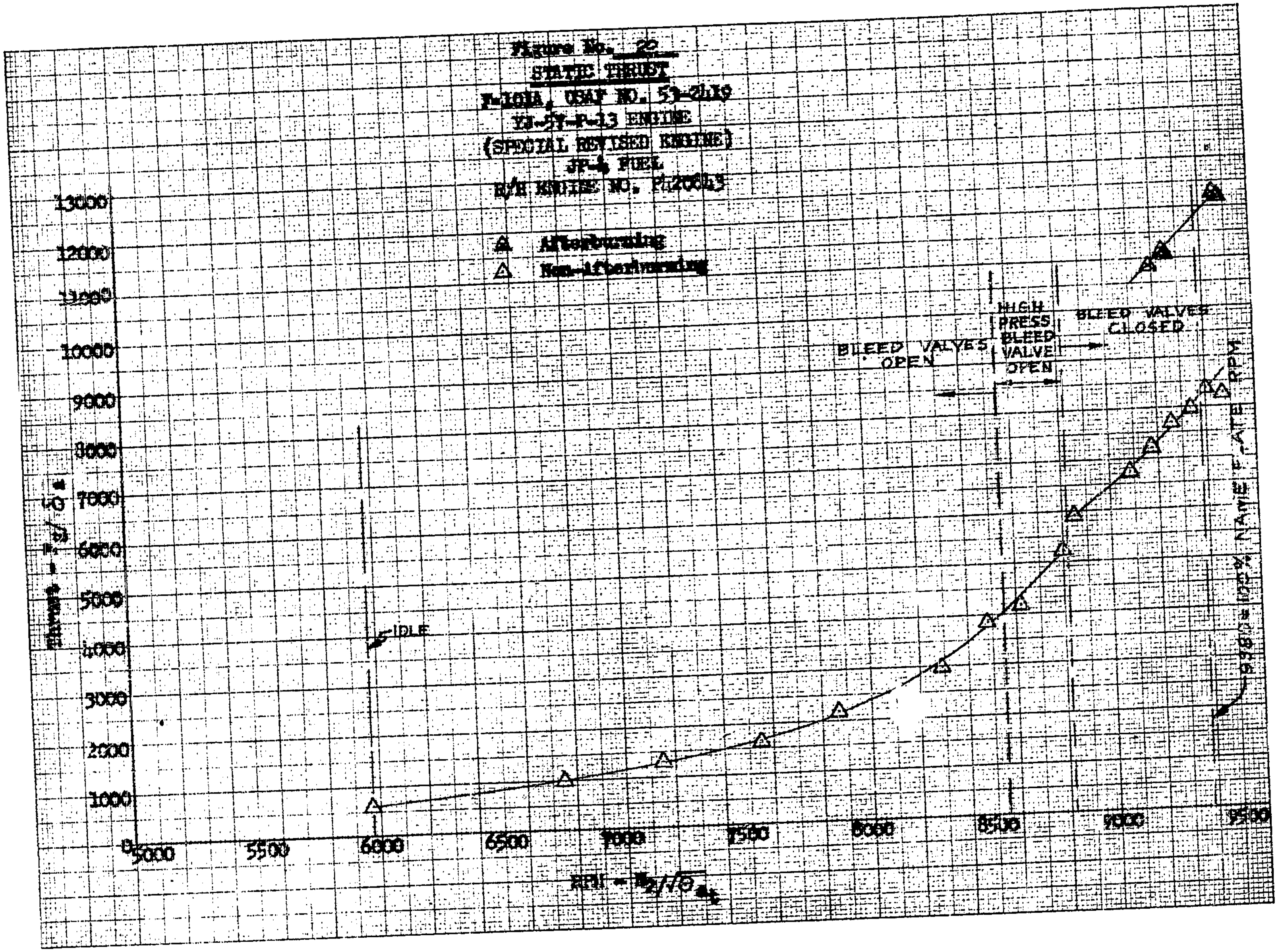


Figure No. 22
STATIC THRUST
F-101A, USAF NO. 53-2419
YJ-57-P-13 ENGINE
(SPECIAL REVISED ENGINE)
JP-4 FUEL
P/W ENGINE NO. P420643
Afterburning
Non-Afterburning
HIGH PRESS BLEED VALVE OPEN
BLEED VALVES OPEN
BLEED VALVES CLOSED
9380=100% NAME PLATE RPM
IDLE
13000
12000
11000
10000
9000
8000
7000
6000
5000
4000
3000
2000
1000
0
5000
5500
6000
6500
7000
7500
8000
8500
9000
9500
RPM - $N_2/\sqrt{\theta_{t_2}}$
Thrust - F_n/δ_{t_2}

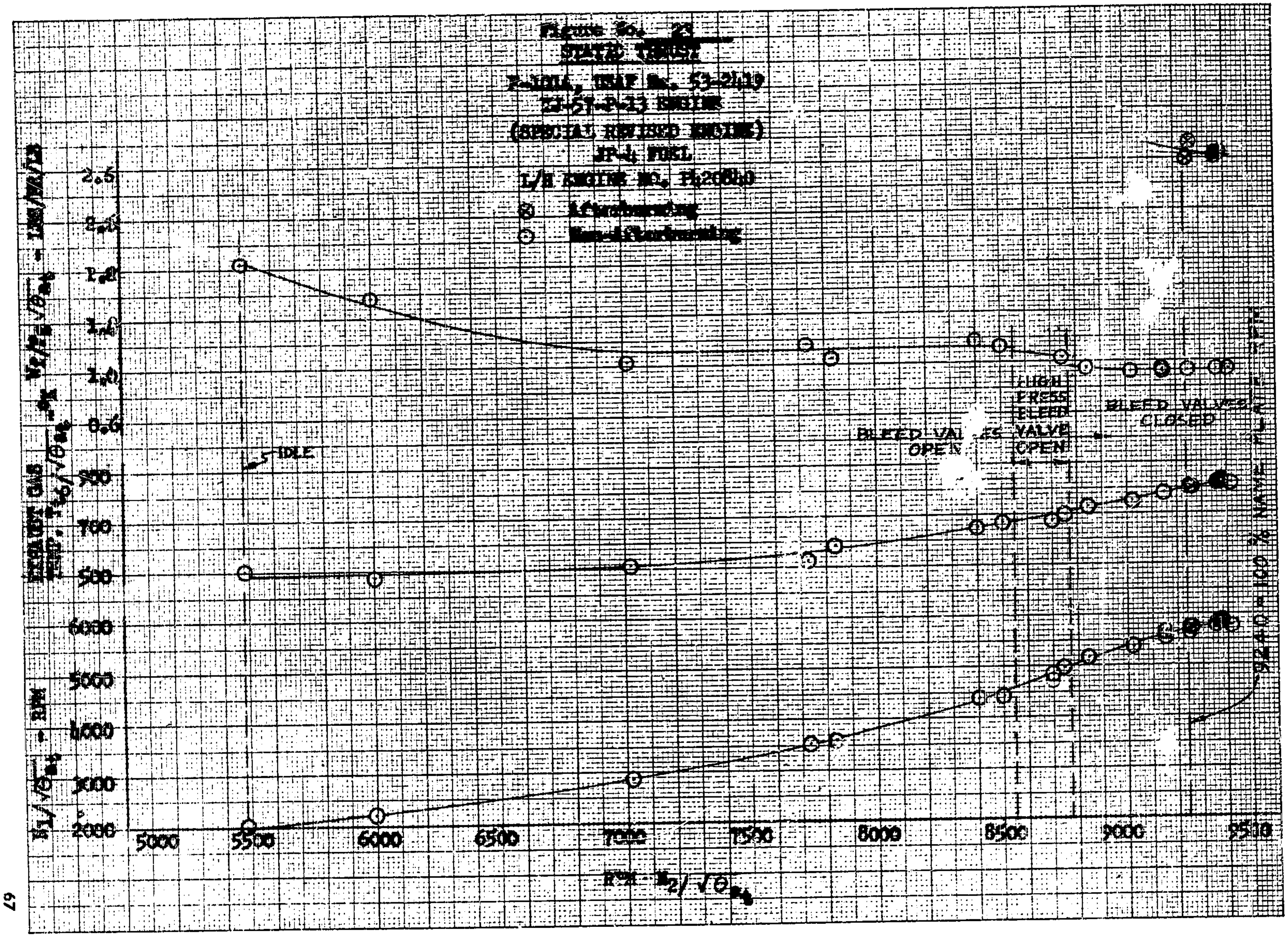


Figure No. 23
STATIC THRUST
F-101A, USAF No. 53-2419
J-57-P-13 ENGINE
(SPECIAL REVISED ENGINE)
JP-4 FUEL
L/H ENGINE NO. P420840
Afterburning
Non-Afterburning
IDLE
BLEED VALVES OPEN
HIGH PRESS BLEED VALVE OPEN
BLEED VALVES CLOSED
9240 = 100 % NAME PLATE RPM
RPM - $N_2/\sqrt{\theta_{t_2}}$
$W_f/F_n/\sqrt{\theta_{t_2}}$ - LBS/HR/LB
EXHAUST GAS TEMP. T_{t_5}/θ_{t_2} - °K
$N_1/\sqrt{\theta_{t_2}}$ - RPM

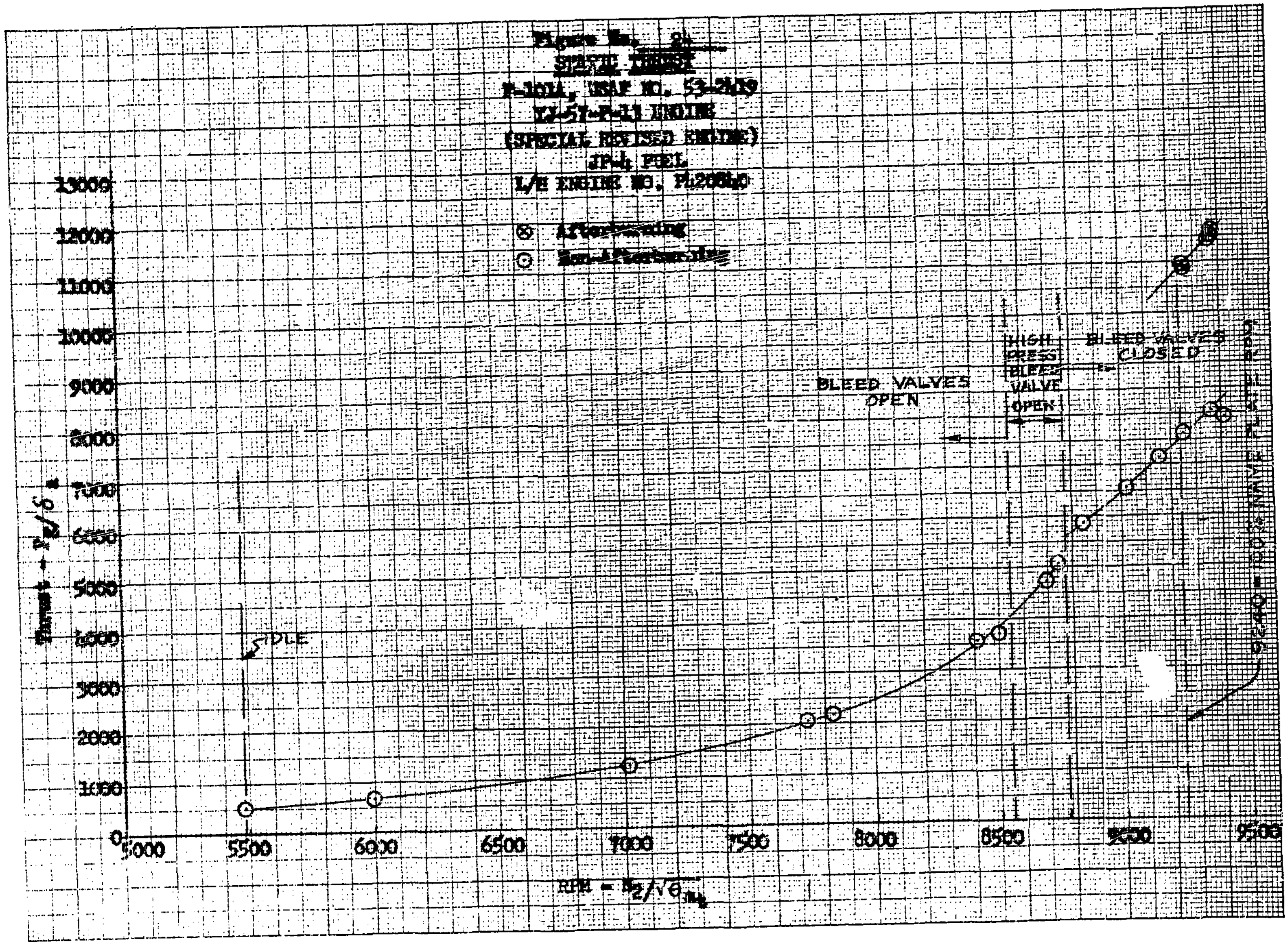


Figure No. 24
STATIC THRUST
F-101A, USAF NO. 53-2419
YJ-57-P-13 ENGINE
(SPECIAL REVISED ENGINE)
JP-4 FUEL
L/H ENGINE NO.
⊗ Afterburning
⊙ Non-Afterburning
HIGH PRESS BLEED VALVE OPEN
BLEED VALVES CLOSED
BLEED VALVES OPEN
IDLE
9240 = 100% NAME PLATE RPM
Thrust - F_n/δ
RPM - $N_2/\sqrt{\theta}$
0
1000
2000
3000
4000
5000
6000
7000
8000
9000
10000
11000
12000
13000
5000
5500
6000
6500
7000
7500
8000
8500
9000
9500

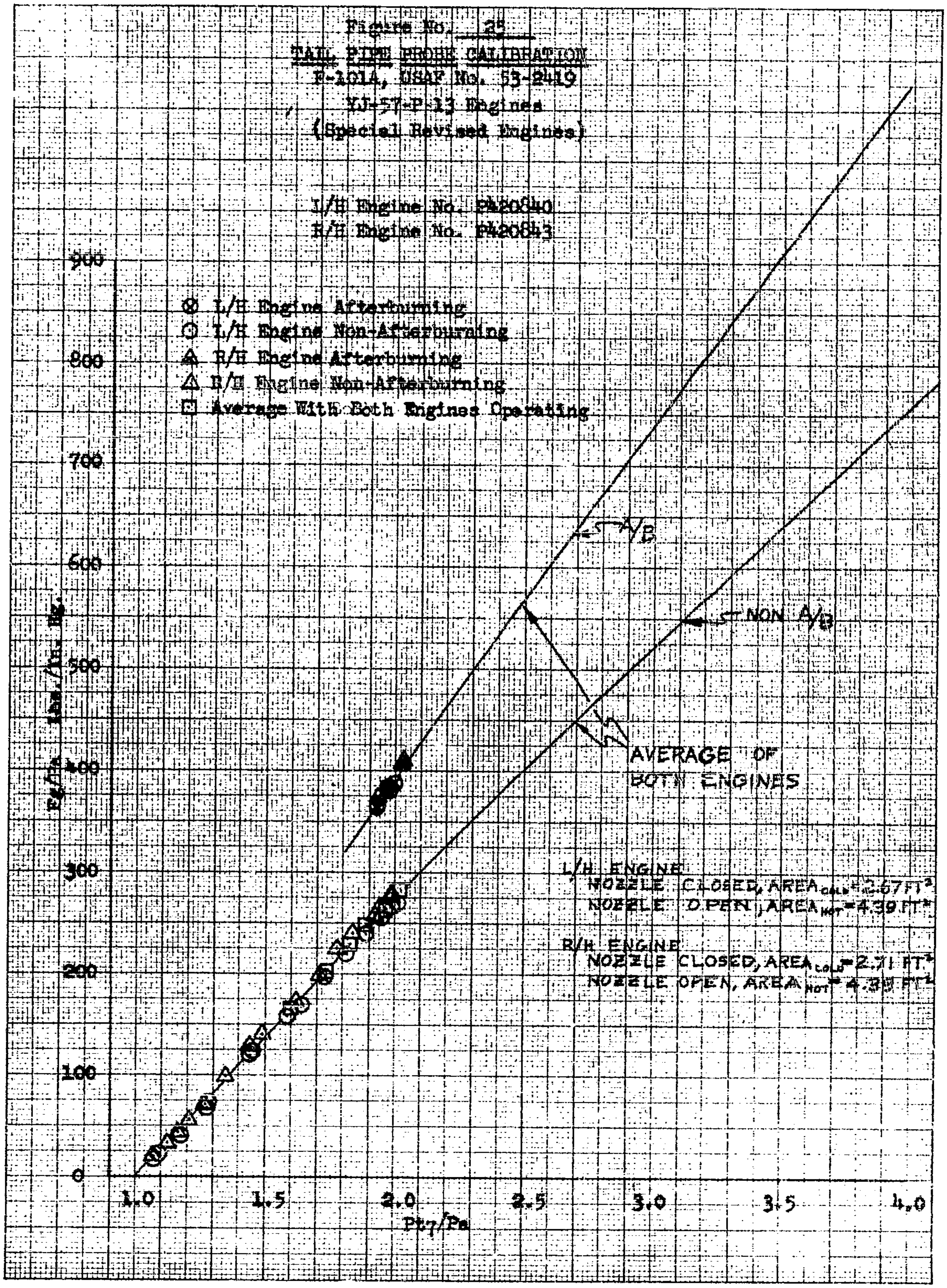


Figure No. 25
TAIL PIPE PROBE CALIBRATION
F-101A, USAF No. 53-2419
XJ-57-P-13 Engines
(Special Revised Engines)
L/H Engine No. P420840
R/H Engine No. P420843
L/H Engine Afterburning
L/H Engine Non-Afterburning
R/H Engine Afterburning
R/H Engine Non-Afterburning
Average With Both Engines Operating
A/B
NON A/B
AVERAGE OF BOTH ENGINES
L/H ENGINE
NOZZLE CLOSED, AREA COLD
NOZZLE OPEN, AREA HOT = 4.39 FT²
R/H ENGINE
NOZZLE CLOSED, AREA COLD = 2.71 FT²
NOZZLE OPEN, AREA HOT
0
100
200
300
400
500
600
700
800
900
1.0
1.5
2.0
2.5
3.0
3.5
4.0
Pt7/Pa

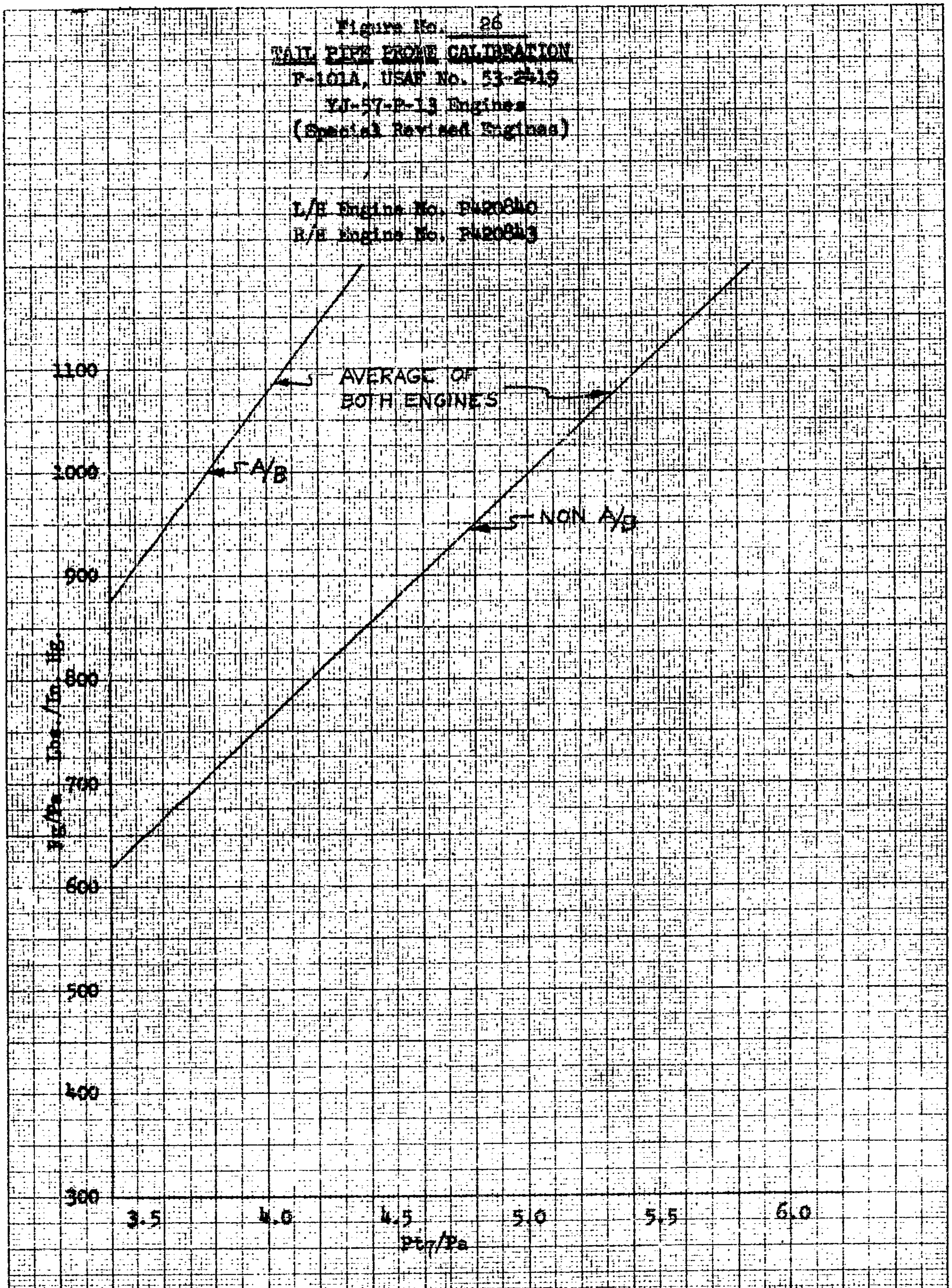
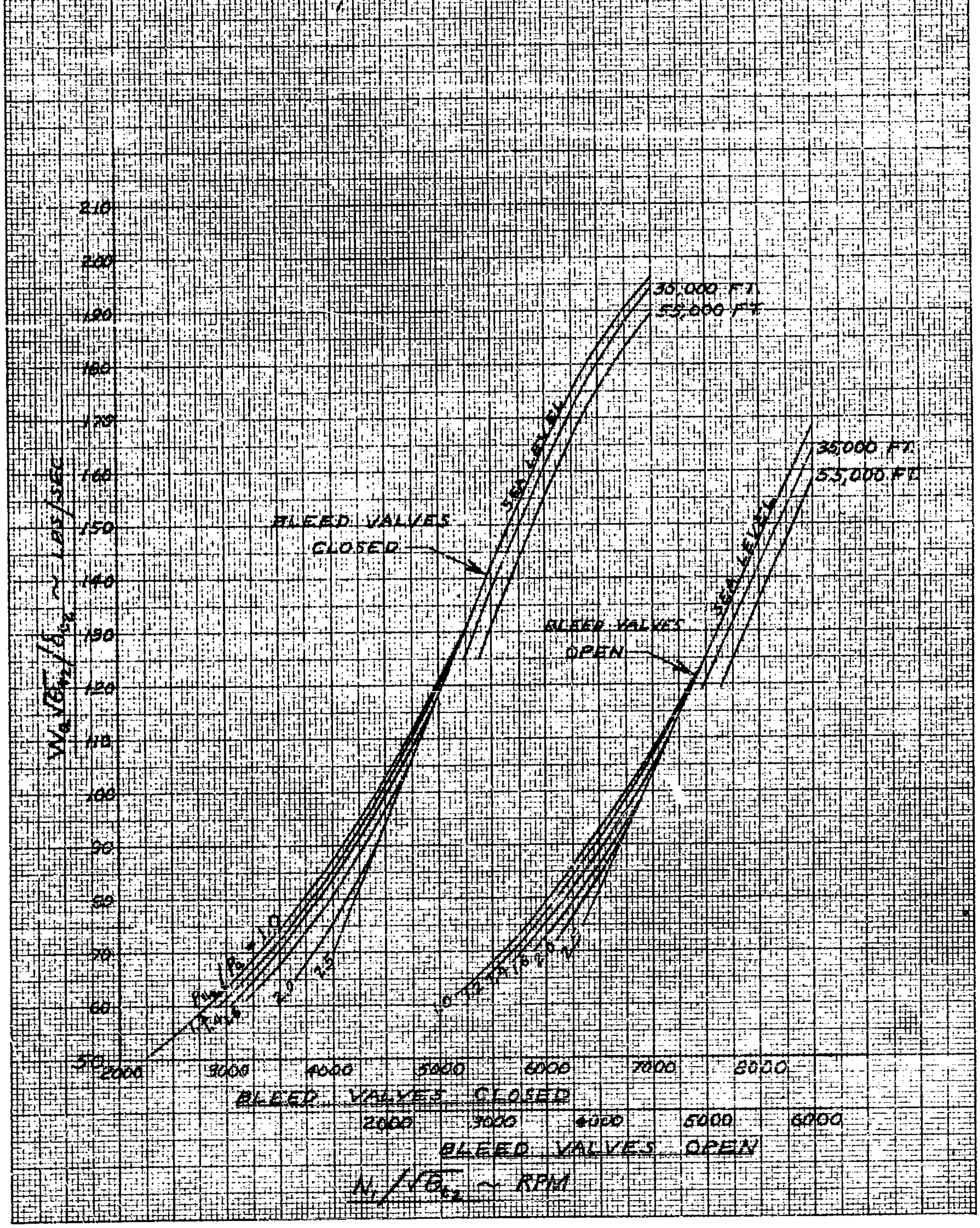


Figure No. 26
TAIL PIPE PROBE CALIBRATION
F-101A, USAF No. 53-2419
YJ-57-P-13 Engines
(Special Revised Engines)
L/H Engine No. P420840
R/H Engine No. P420843
AVERAGE OF BOTH ENGINES
A/B
NON A/B
1100
1000
900
800
700
600
500
400
300
3.5
4.0
4.5
5.0
5.5
6.0
Pt7/Pa

FIG. No 27
CORRECTED AIR FLOW
F-101A USAF No. 53-2419
XJ-57-P-13 ENGINE



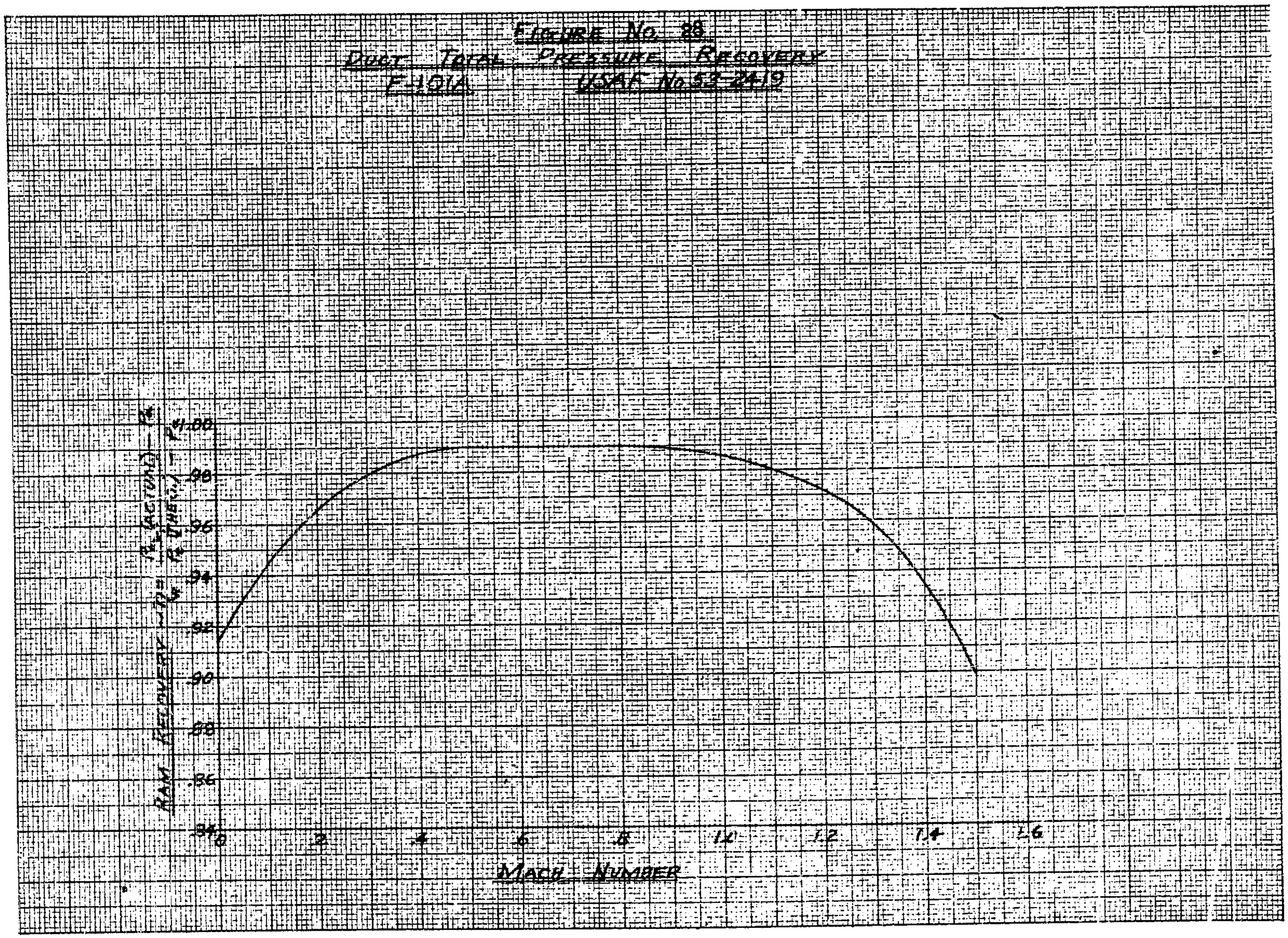


FIGURE No. 28
DUCT TOTAL PRESSURE RECOVERY
F-101A USAF No. 53-2419

stability and control plots

In the plots marked with an asterisk, the aircraft had the total pressure sensing feel system pick-up located on the leading edge of the vertical stabilizer. In all other plots, the aircraft had the total pressure sensing feel system pick-up located on the left side of the vertical stabilizer.

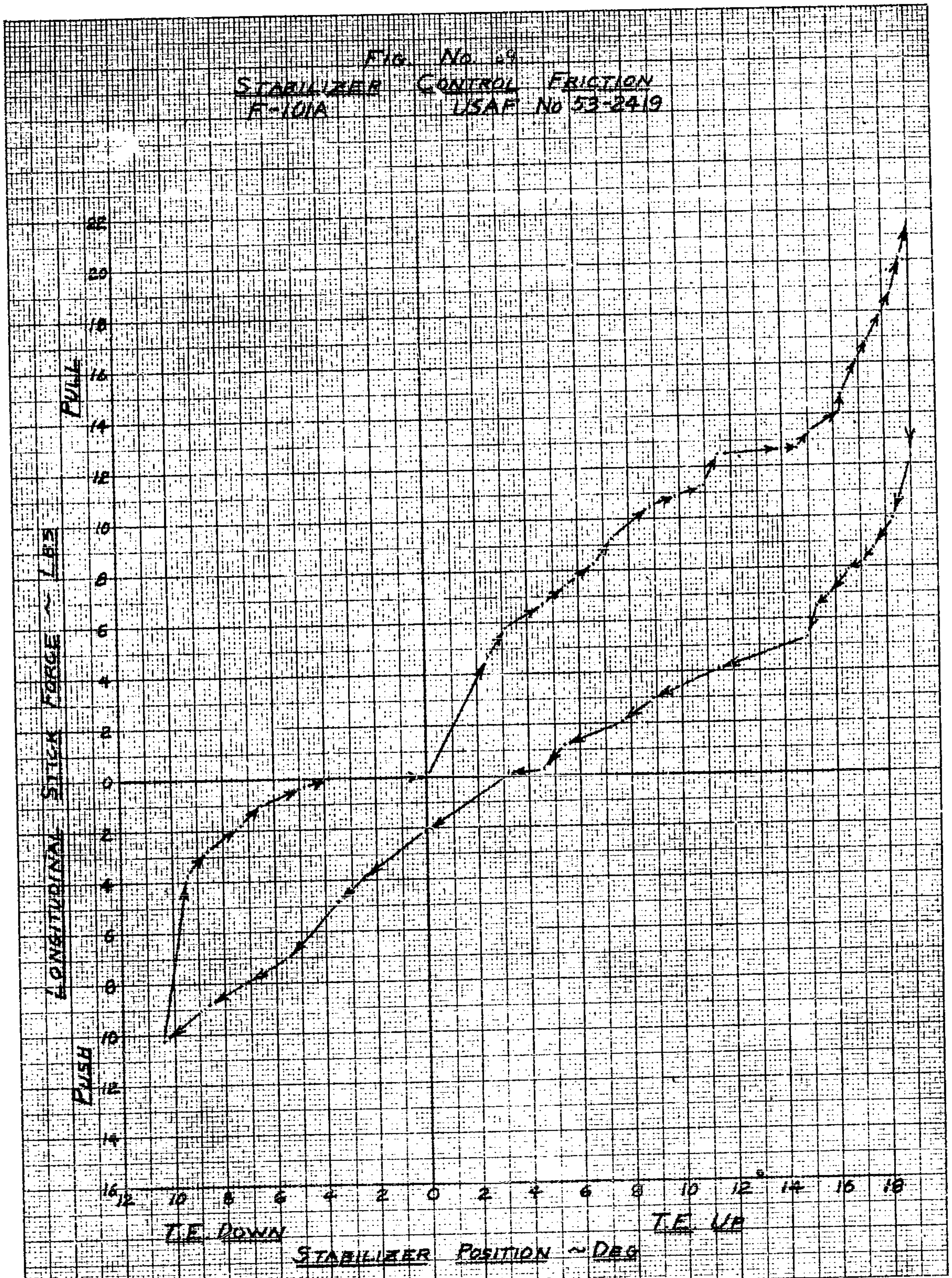


FIG. NO. 29
STABILIZER CONTROL FRICTION
F-101A USAF No 53-2419
PULL
LONGITUDINAL STICK FORCE ~ LBS
PUSH
22
20
18
16
14
12
10
8
6
4
2
0
2
4
6
8
10
12
14
16
12
10
8
6
4
2
0
2
4
6
8
10
12
14
16
18
T.E. DOWN
T.E. UP
STABILIZER POSITION ~ DEG

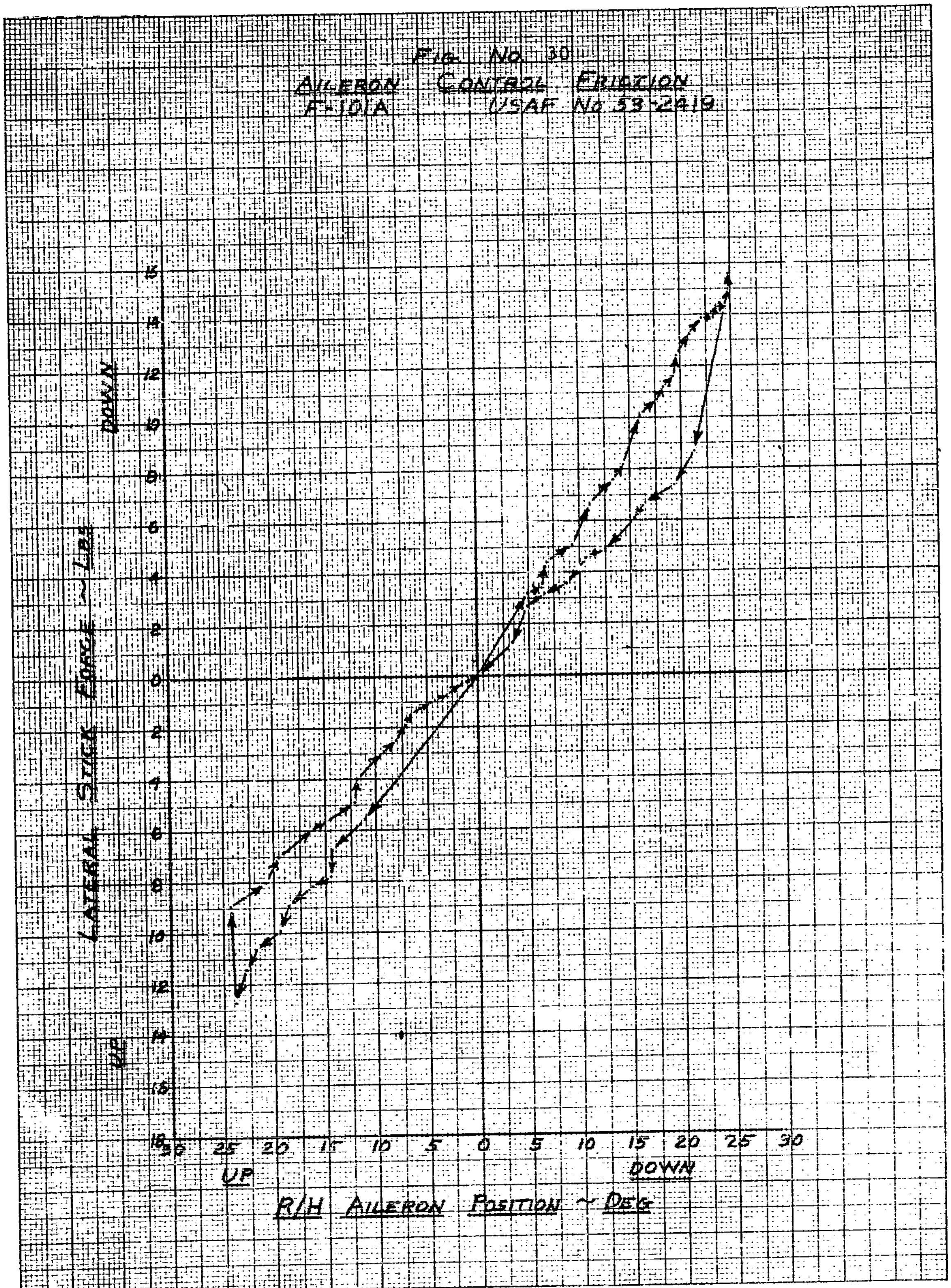


FIG. No. 30
AILERON CONTROL FRICTION
F-101A USAF No 53-2419
LATERAL STICK FORCE ~ LBS
DOWN
UP
16
14
12
10
8
6
4
2
0
2
4
6
8
10
12
14
16
18
30 25 20 15 10 5 0 5 10 15 20 25 30
UP
DOWN
R/H AILERON POSITION ~ DEG

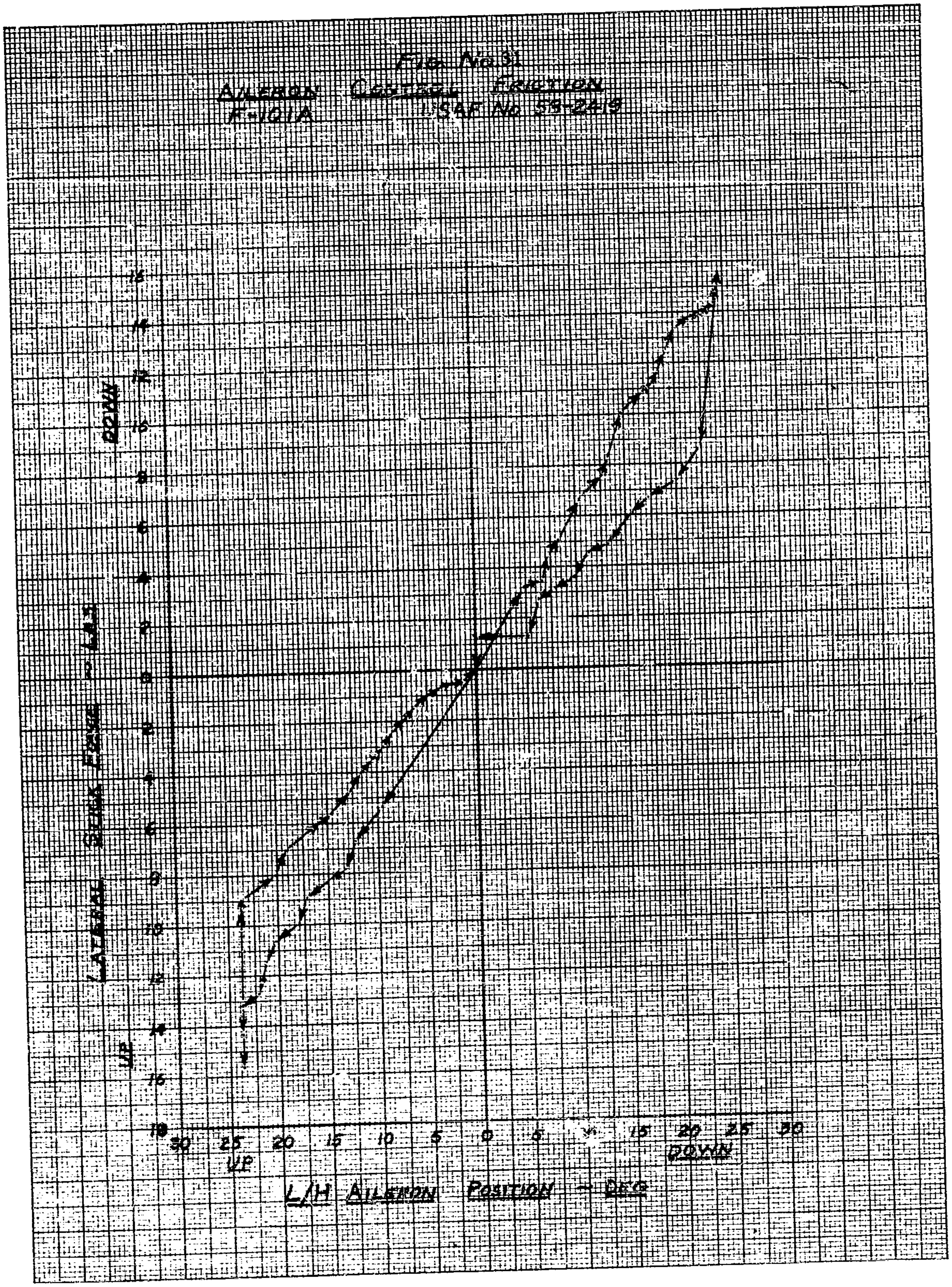


FIG. No. 31
AILERON CONTROL FRICTION
F-101A USAF No. 53-2419
LATERAL STICK FORCE ~ LBS.
DOWN
UP
16
14
12
10
8
6
4
2
0
2
4
6
8
10
12
14
16
18
30
25
20
15
10
5
0
5
15
20
25
30
UP
DOWN
L/H AILERON POSITION ~ DEG

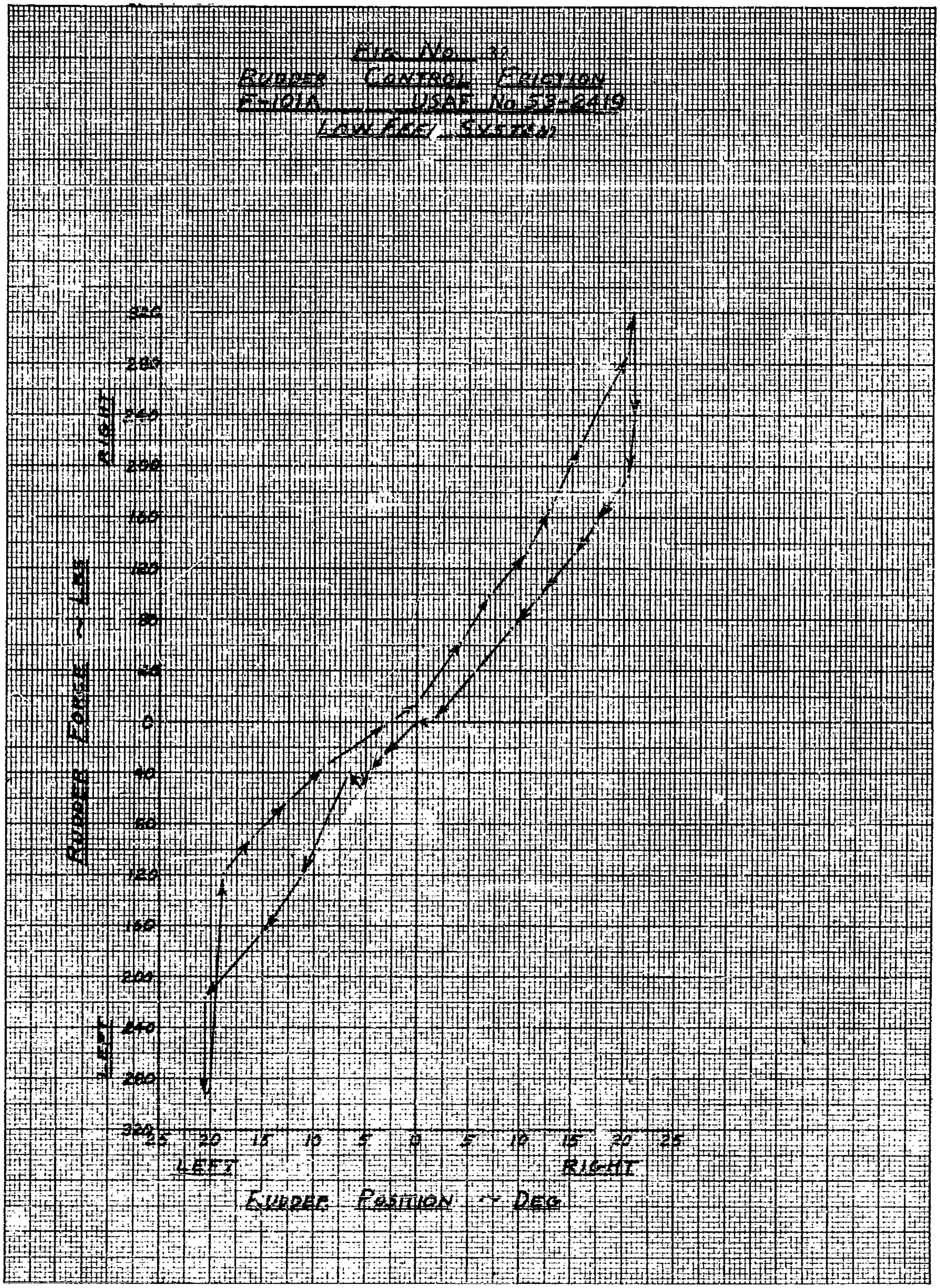


FIG. NO.
RUDDER CONTROL FRICTION
F-101A USAF No 53-2419
LOW FEEL SYSTEM
RUDDER FORCE ~ LBS
RIGHT
LEFT
320
280
240
200
160
120
80
40
0
40
80
120
160
200
240
280
320
25 20 15 10 5 0 5 10 15 20 25
LEFT
RIGHT
RUDDER POSITION ~ DEG

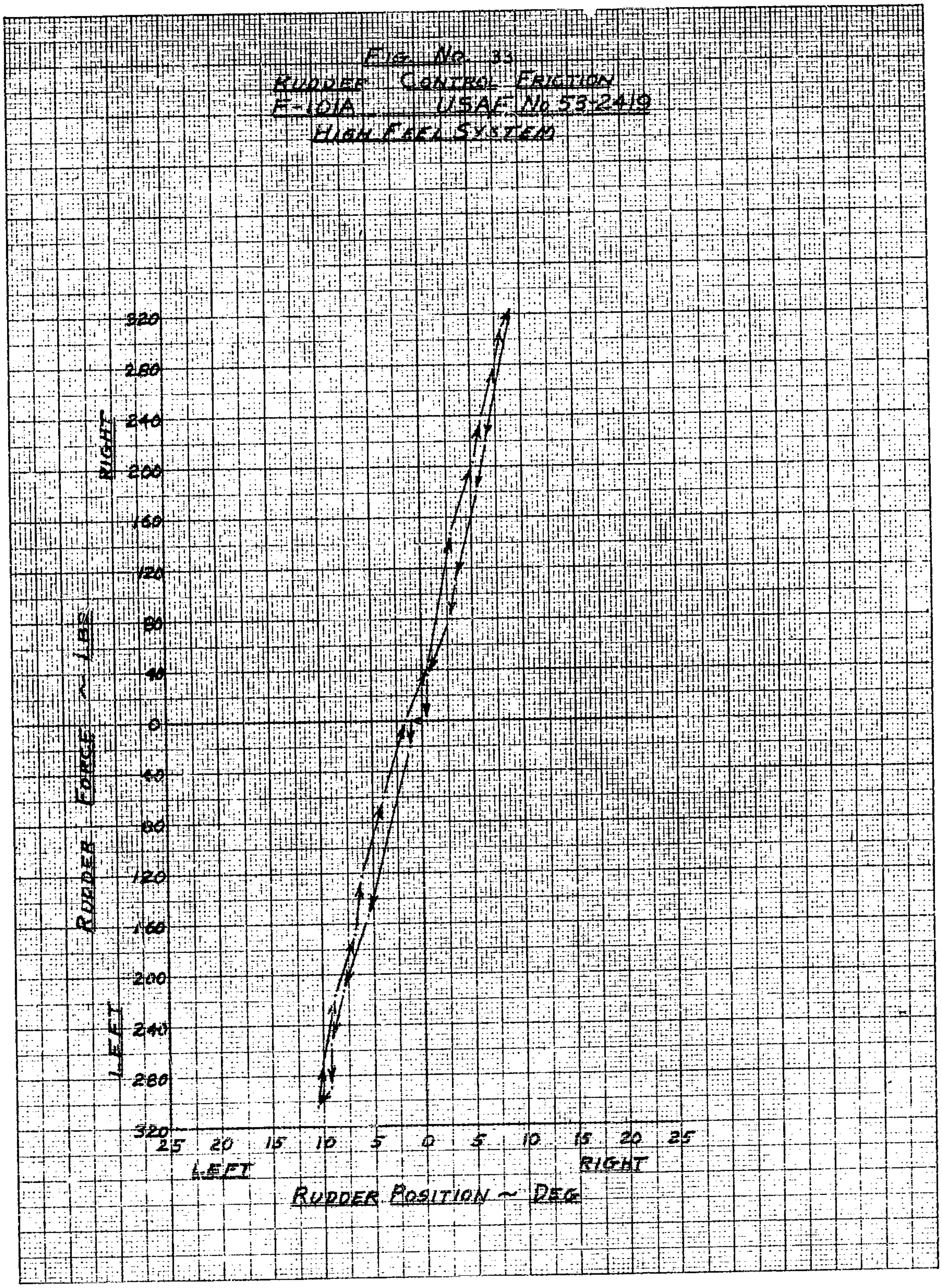


FIG. NO. 33
RUDDER CONTROL FRICTION
F-101A USAF No 53-2419
HIGH FEEL SYSTEM
RUDDER FORCE ~ LBS
RIGHT
LEFT
320
280
240
200
160
120
80
40
0
40
80
120
160
200
240
280
320
25
20
15
10
5
0
5
10
15
20
25
LEFT
RIGHT
RUDDER POSITION ~ DEG

Figure No. 34

TAKE-OFF TIME HISTORY*

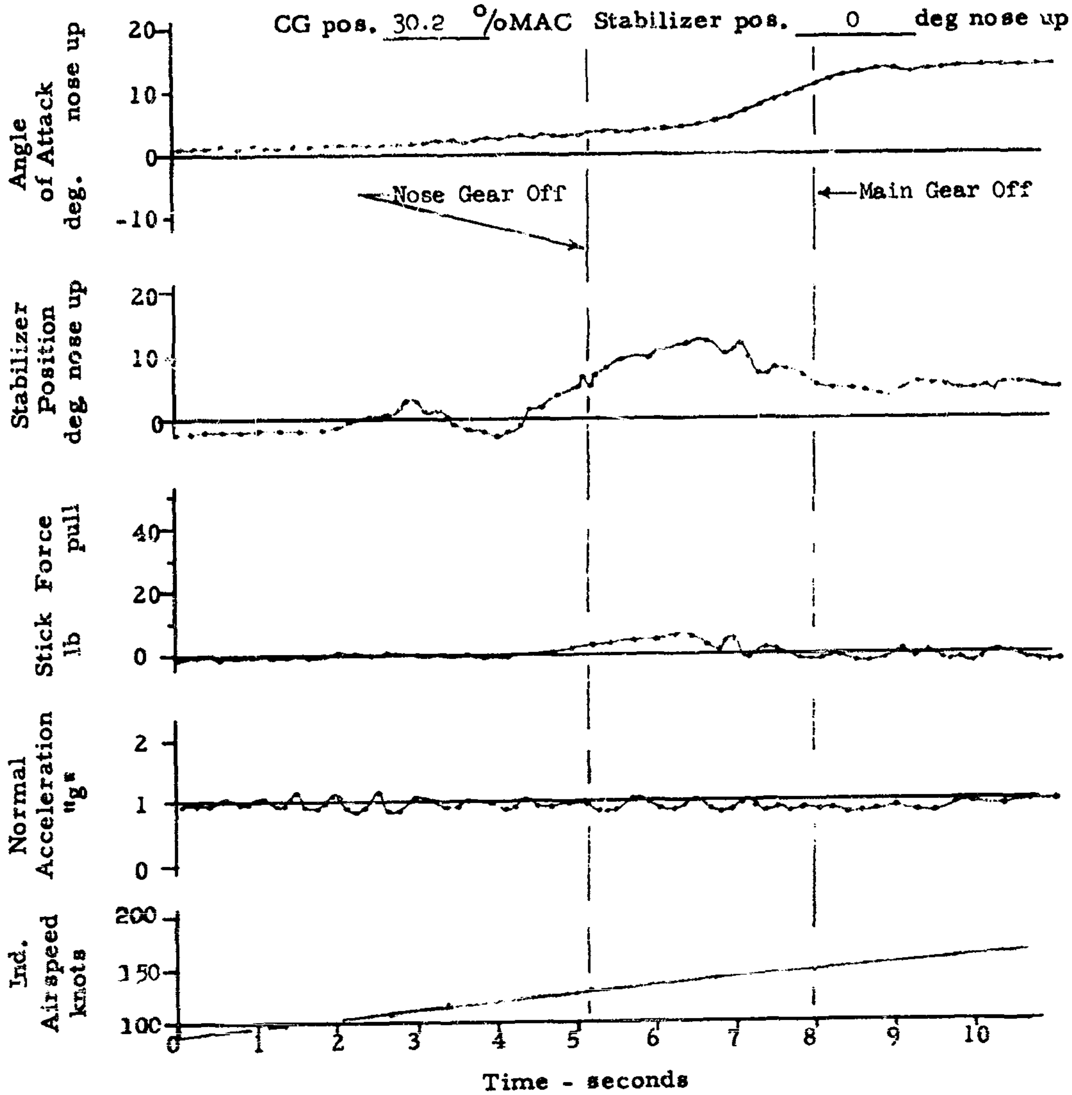
F-101A, USAF No. 53-2419

Take-Off Configuration

TRIM CONDITIONS

Alt. 2160 feet Weight 40,370 lb

CG pos. 30.2 %MAC Stabilizer pos. 0 deg nose up



APPENDIX I

Figure No. 35

TAKE-OFF TIME HISTORY*

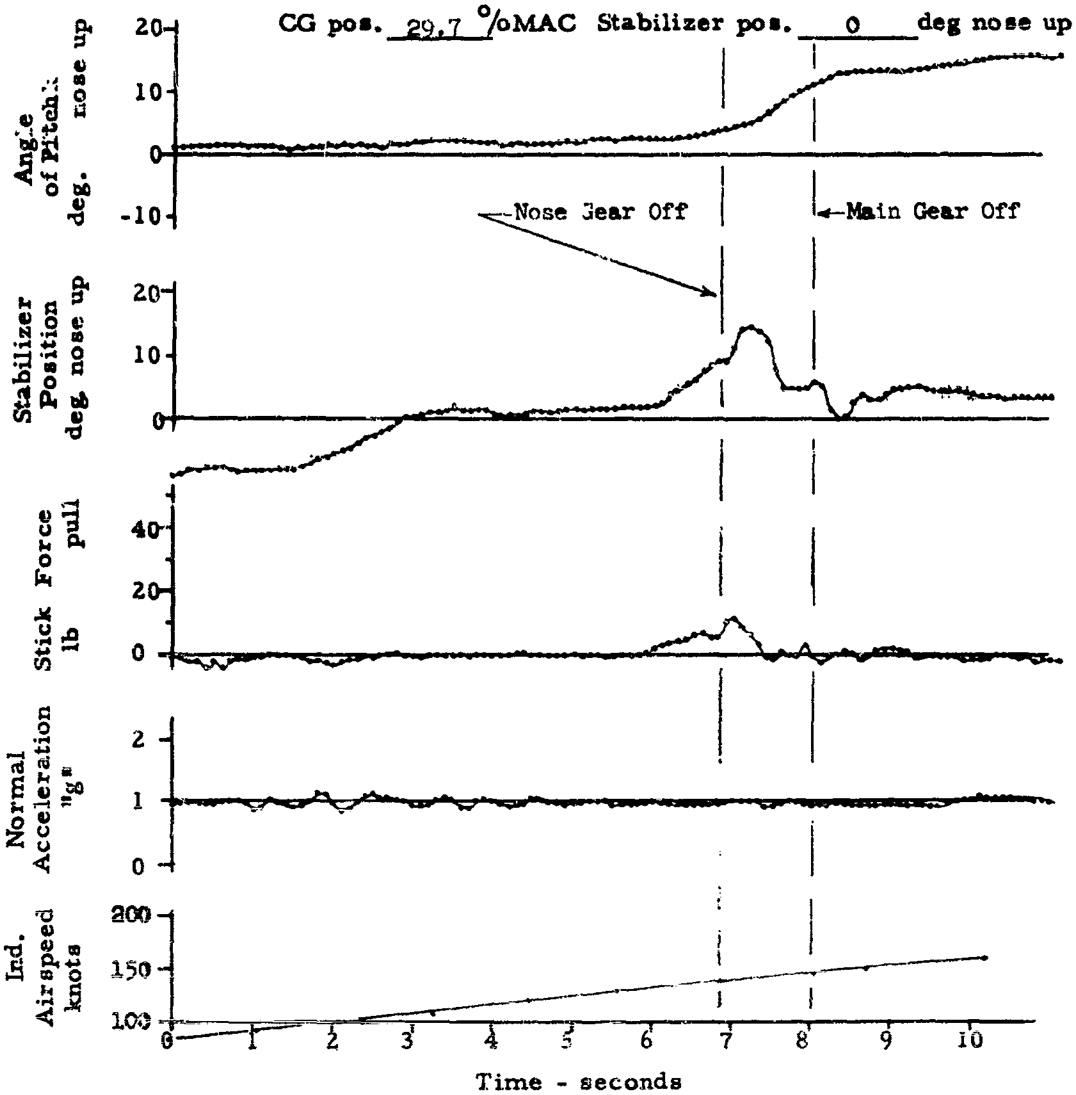
F-101A, USAF No. 53-2419

Take-Off Configuration

TRIM CONDITIONS

Alt. 2150 feet Weight 40,460 lb

CG pos. 29.7 %MAC Stabilizer pos. 0 deg nose up



APPENDIX I

Figure No. 36

TAKE-OFF TIME HISTORY*

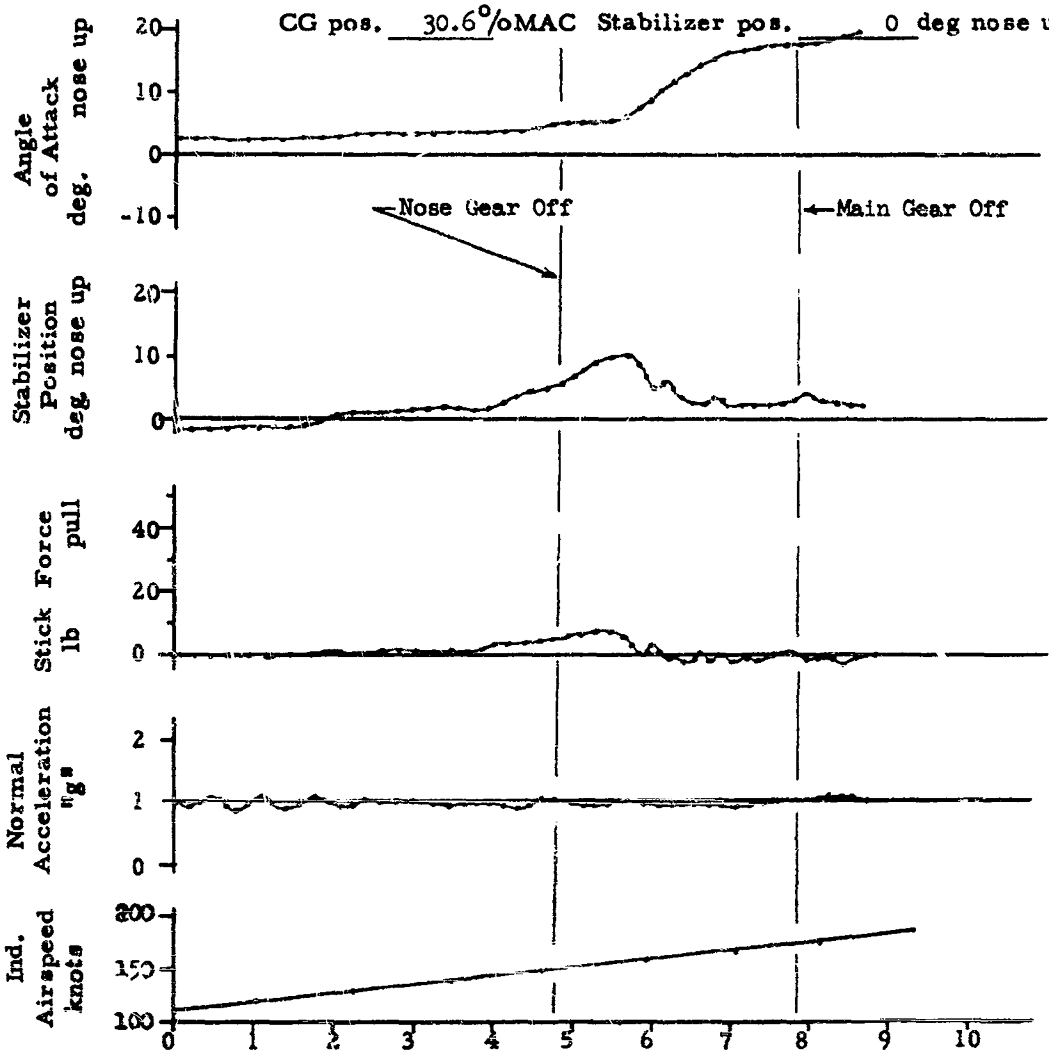
F-101A, USAF No. 53-2419

Take-Off Configuration

TRIM CONDITIONS

Alt. 2140 feet Weight 40,410 lb

CG pos. 30.6 %MAC Stabilizer pos. 0 deg nose up

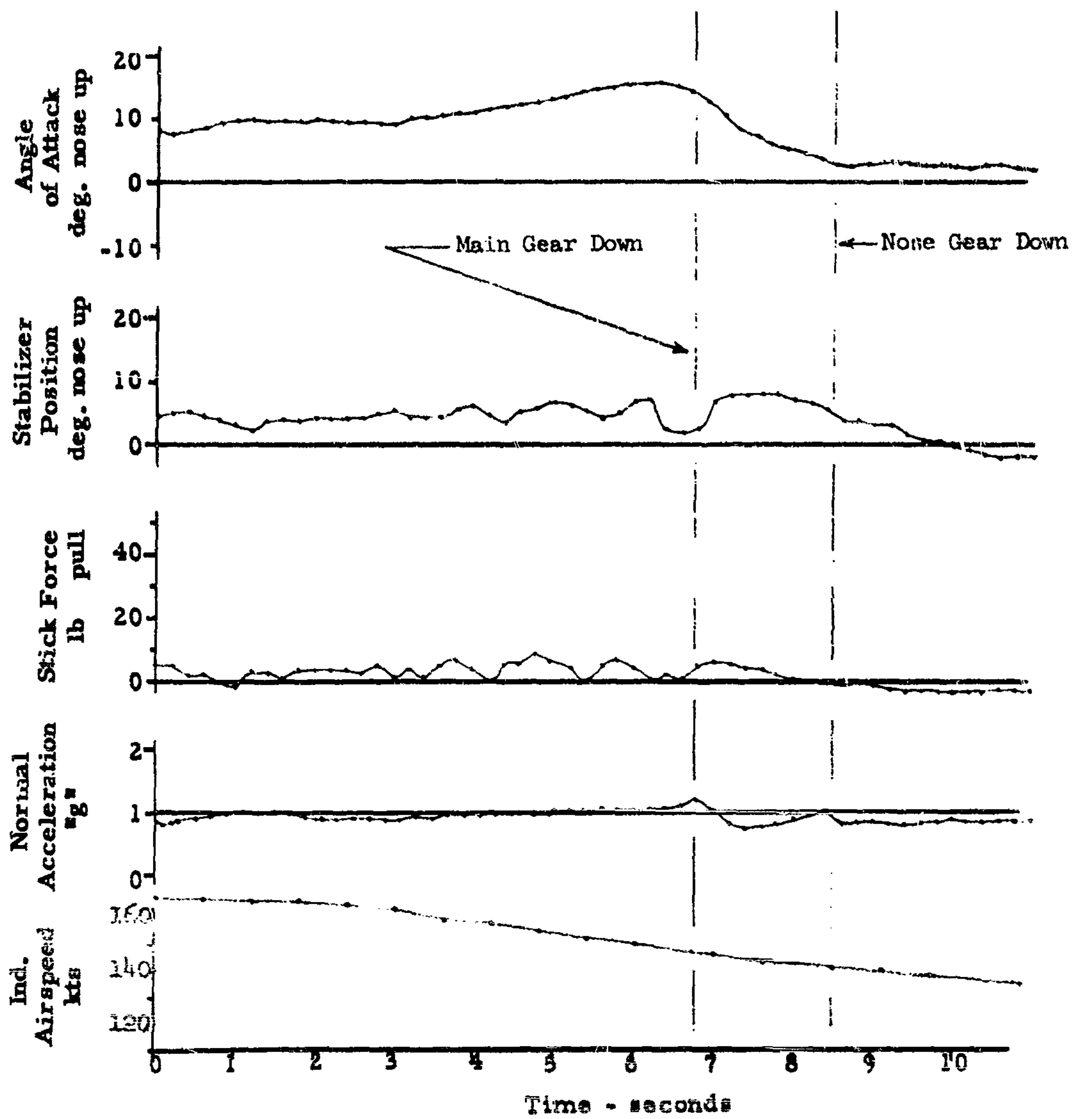


APPENDIX I

Figure No. 37
LANDING TIME HISTORY*
F-101A, USAF No. 53-2419
Landing Configuration

TRIM CONDITIONS

Alt. 2180 feet Weight 30,140 lb
CG pos. 33.6 % MAC Stabilizer pos. 3.0 deg. nose up

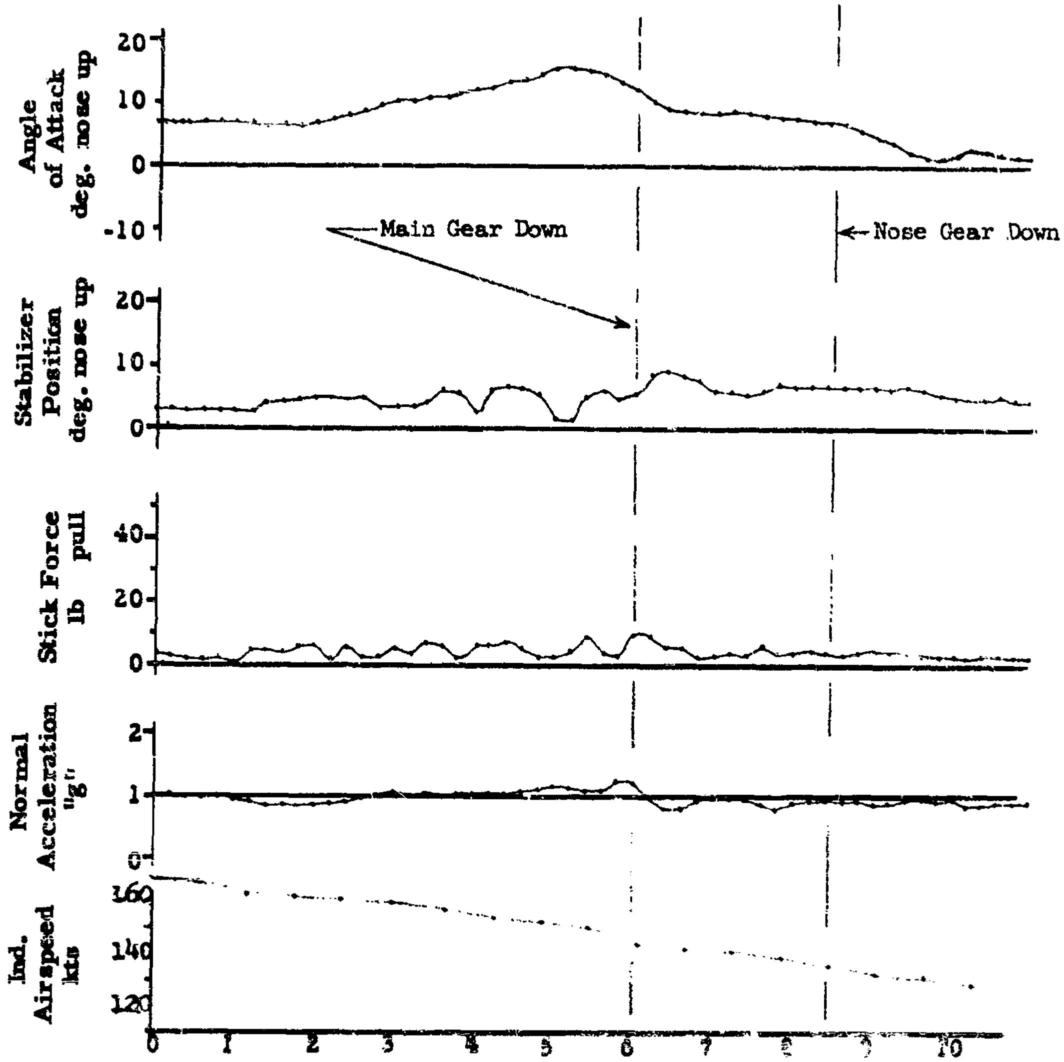


APPENDIX I

Figure No. 38
LANDING TIME HISTORY*
F-101A, USAF No. 53-2419
Landing Configuration

TRIM CONDITIONS

Alt. 2270 feet Weight 31,455 lb
CG pos. 32.9 %MAC Stabilizer pos. 2.0 deg. nose up



APPENDIX I

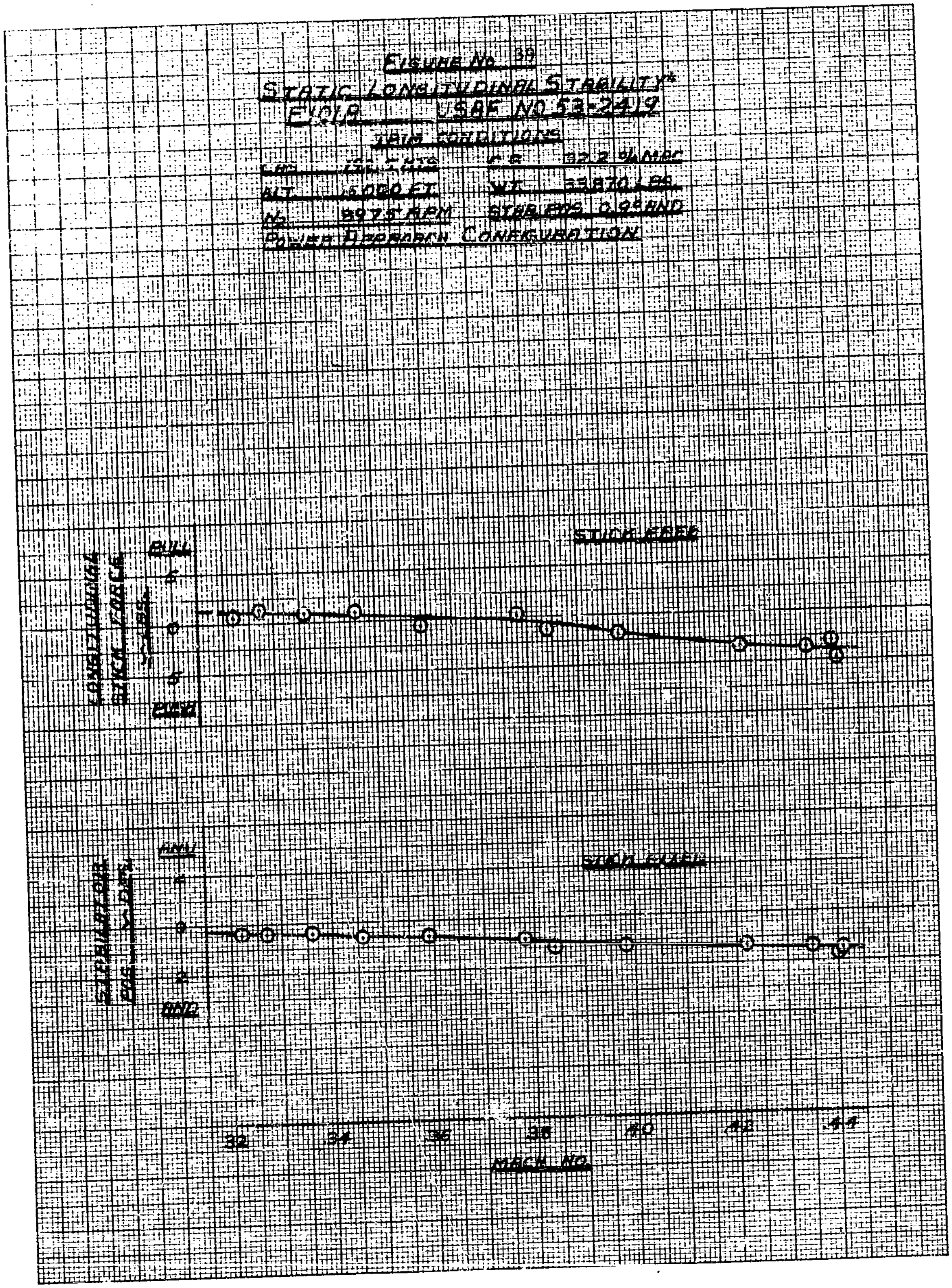


FIGURE NO. 39
STATIC LONGITUDINAL STABILITY
USAF NO 53-2419
TRIM CONDITIONS
WT.
STAB. POS.
CONFIGURATION
STICK FREE
PULL
PUSH
STICK FIXED
AND
.32
.34
.36
.38
.40
.42
.44
MACH NO.

FIGURE NO. 40

STATIC LONGITUDINAL STABILITY

F101B USAF NO 53-2419

TRIM CONDITIONS

CAS [illegible] KTS C.G. [illegible] % MAC

ALT. 18,000 FT. WT. 36770 LBS

N_2 [illegible] RPM STAB POS. 1.0° ANU

CRUISE CONFIGURATION

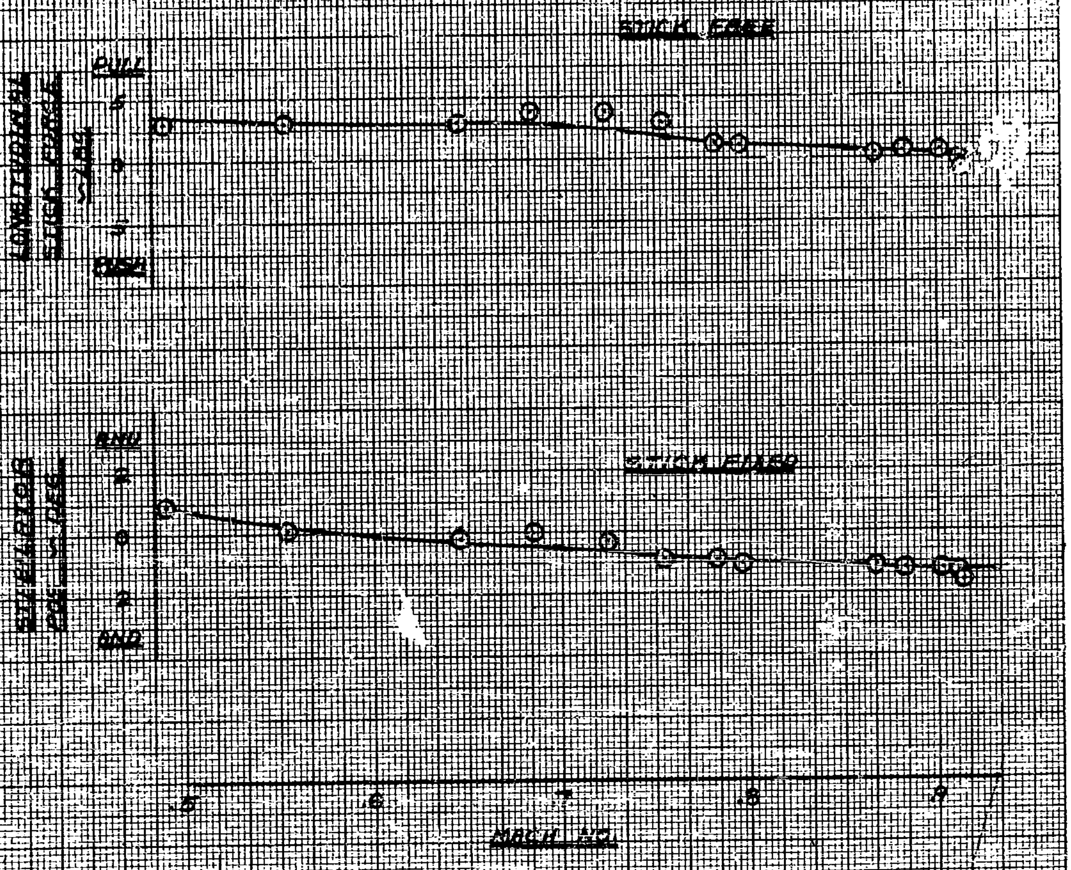


FIGURE NO. 41

STATIC LONGITUDINAL STABILITY

F101A USAF NO. 53-2419

TRIM CONDITIONS

CAS 287 KTS. C.G. 33.2% MAC

ALT. 35000 FT. WT. 38050 LBS

N_2 8275 RPM STAB. POS. 0°

CRUISE CONFIGURATION

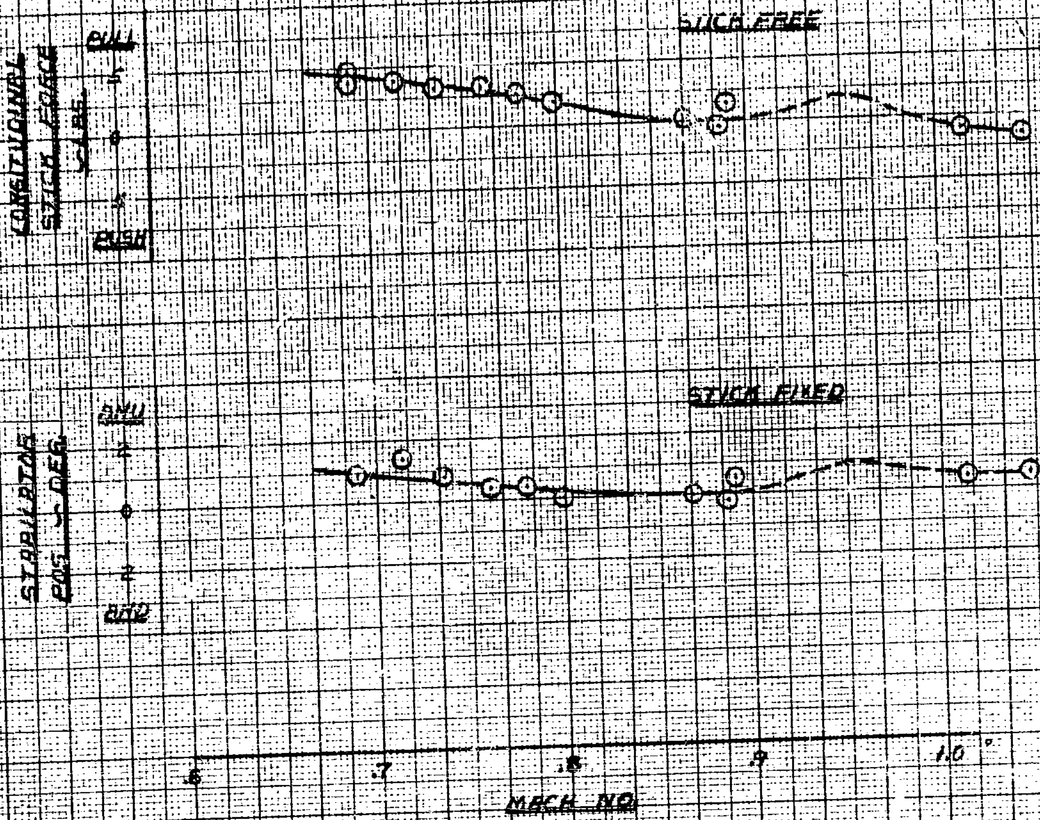


FIGURE NO. 42

STATIC LONGITUDINAL STABILITY

F101A USAF NO. 53-2419

TRIM CONDITIONS

CAS 379.5 KTS. C.G. 33.2 % MAC

ALT. 37000 FT. WT. 37040 LBS.

N_2 9205 RPM STAB. POS. 0.5° ANU

COMBAT CONFIGURATION

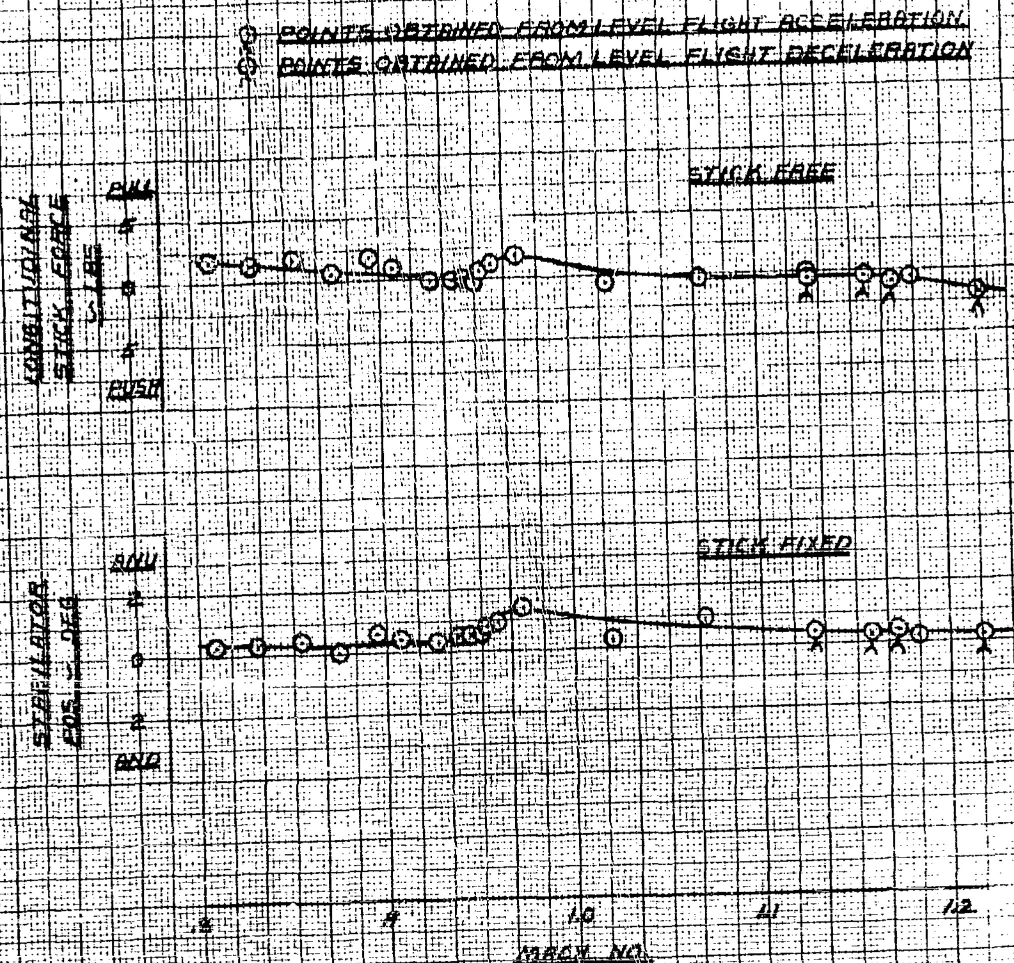


FIGURE NO. 43

STATIC LONGITUDINAL STABILITY

F101A USAF NO. 53-2419

TRIM CONDITIONS

CAS	[illegible] KTS	C.G.	[illegible] % MAC
ALT	[illegible] FT	WT	[illegible] LBS
N_2	9280 RPM	STAB. POS.	0.4° ANU

COMBAT CONFIGURATION

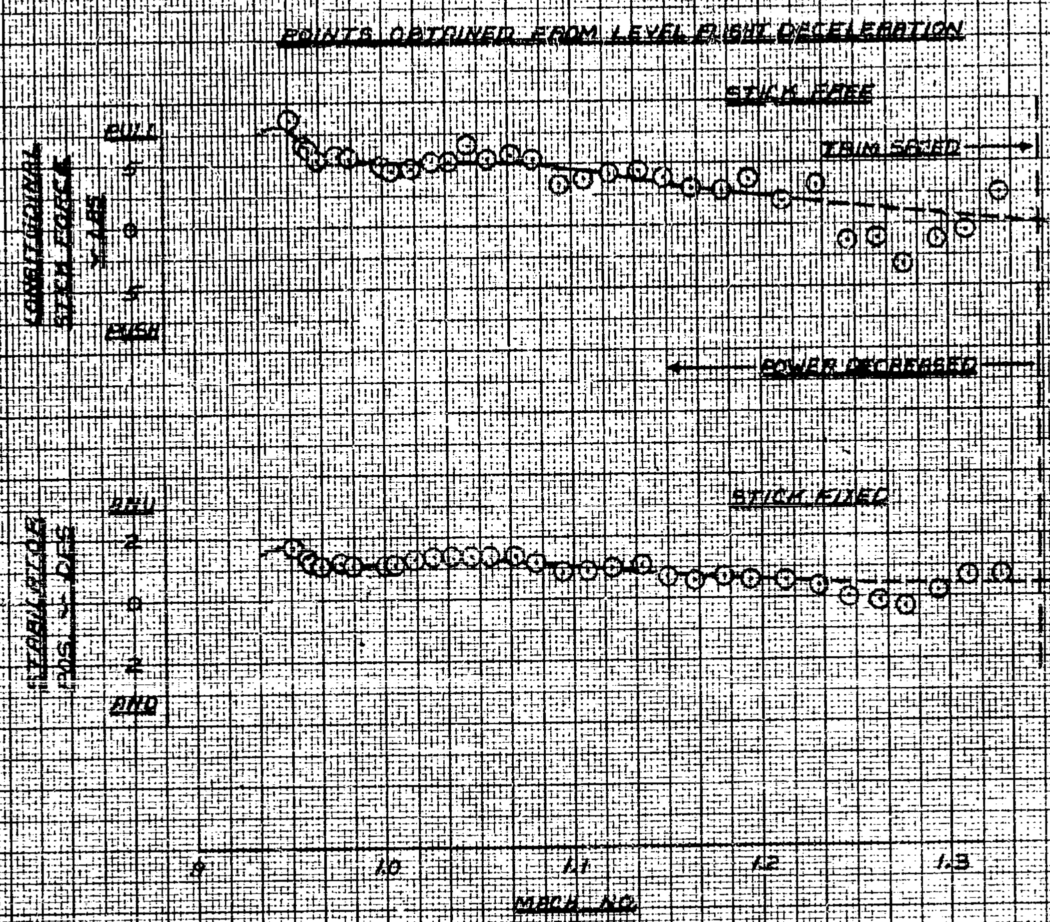


Figure No. 44

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

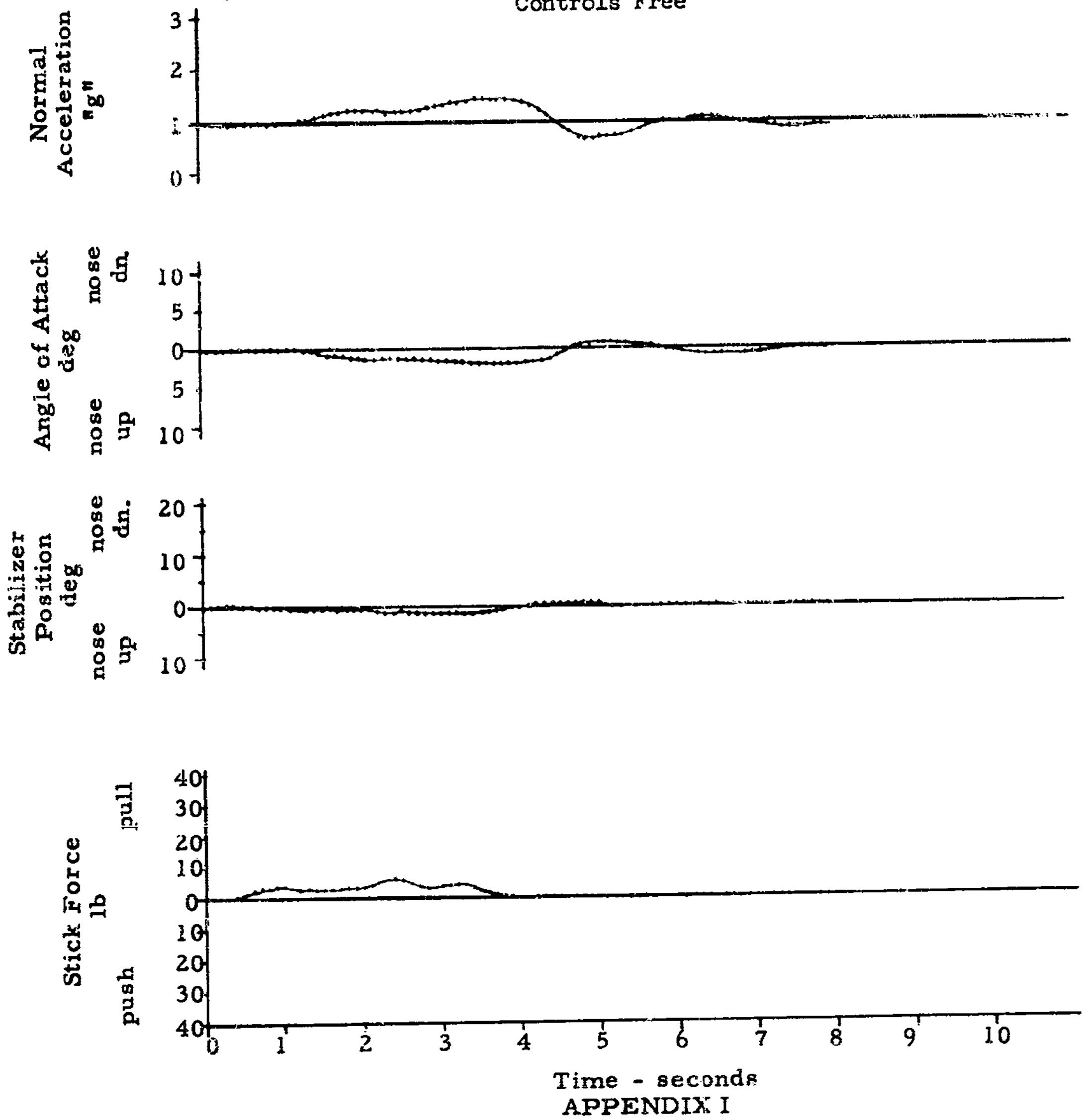
CAS 278.5 kts CG pos. 33.5 % MAC

Alt. 38,400 ft Weight 32,430 lb

Avg. N_2 8290 rpm Stabilizer .3 Deg. Nose Up

Mach No. .877

Controls Free



APPENDIX I

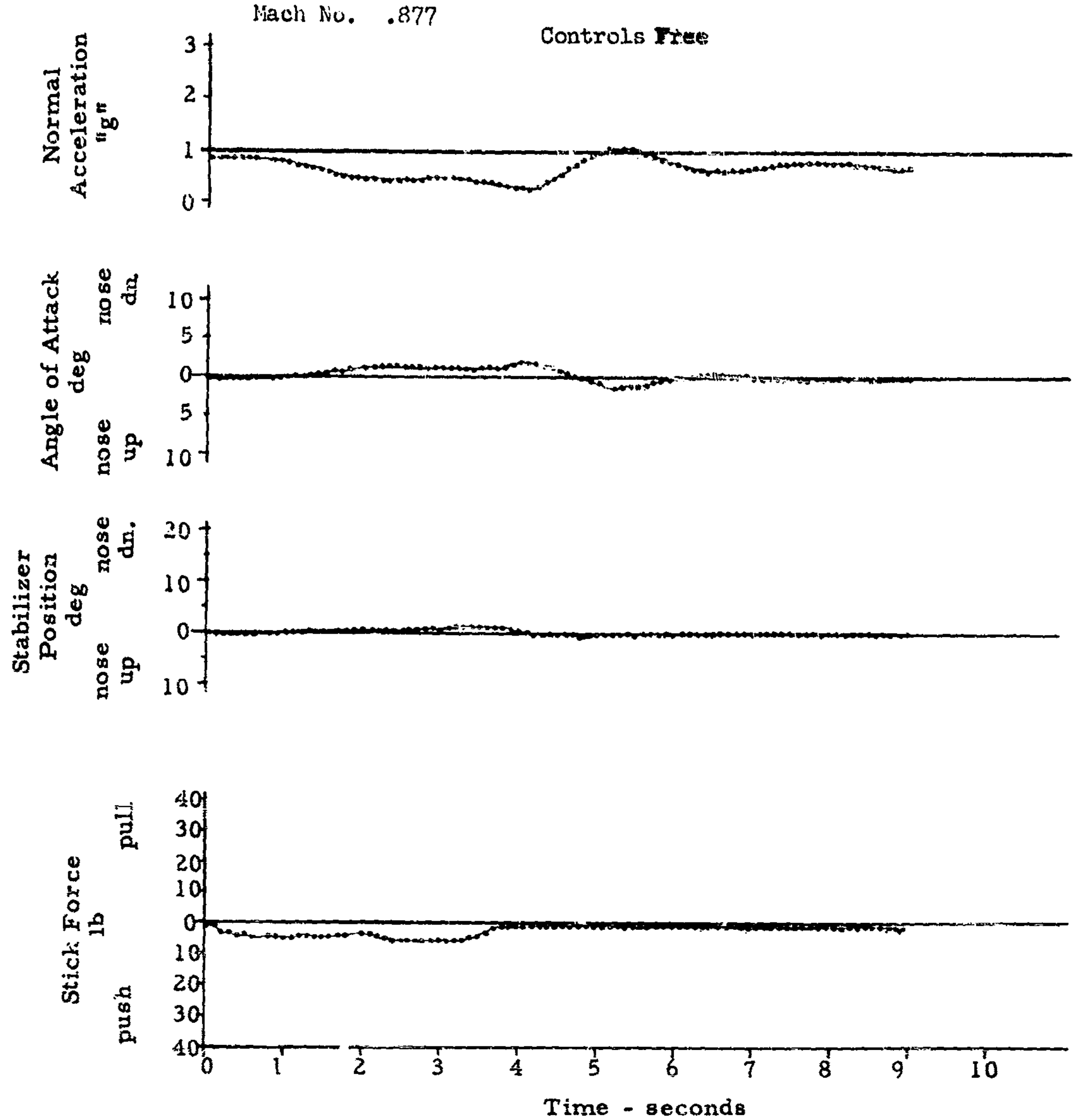
Figure No. 45

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No.53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 278.5 kts CG pos. 33.5 %MAC
Alt. 38,400 ft Weight 32,430 lb
Avg. N_2 8290 rpm Stabilizer .3 Deg. Nose Up
Mach No. .877

Controls Free



APPENDIX I

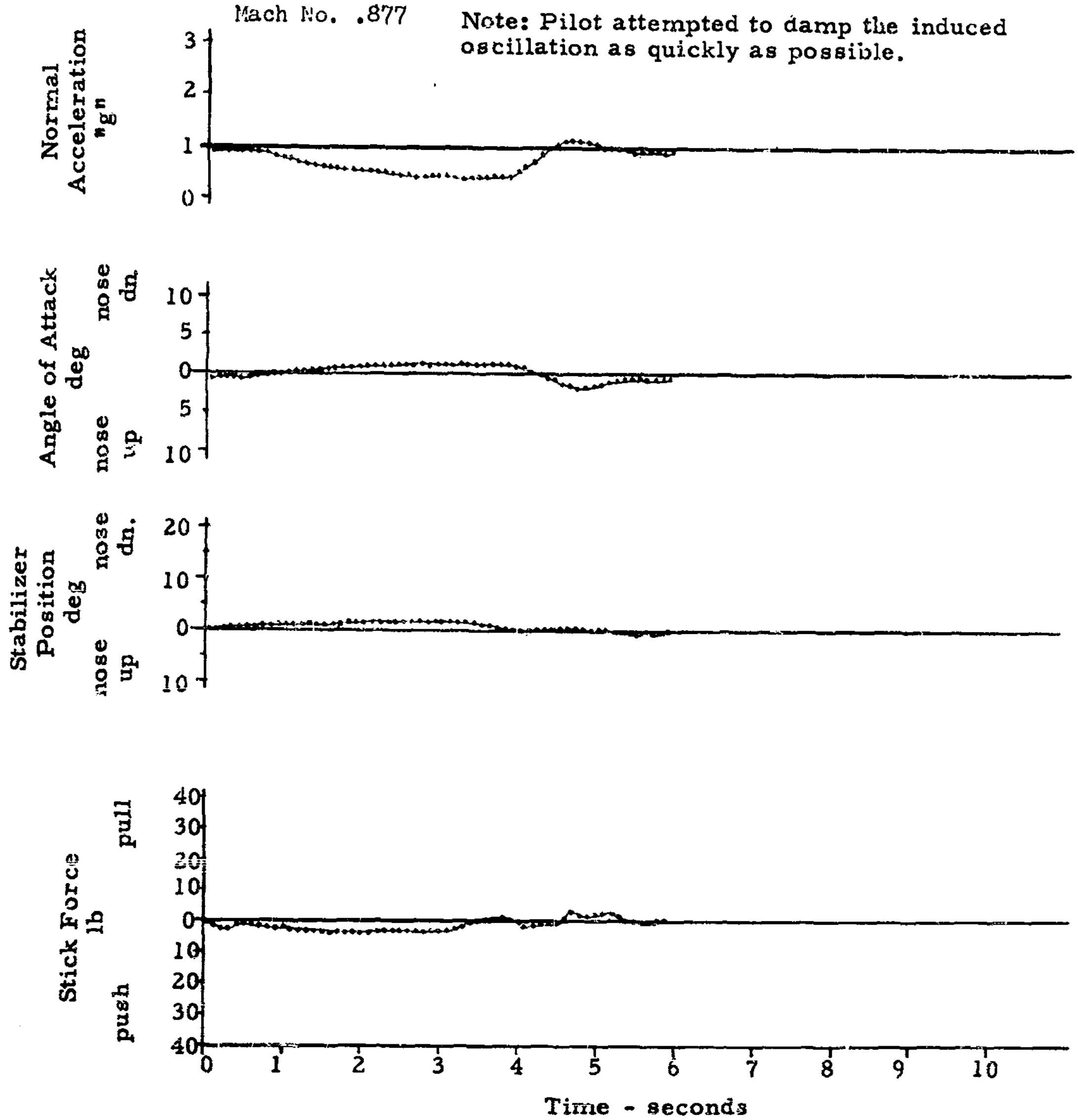
Figure No. 46

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 278.5 kts CG pos. 33.5 %MAC
Alt. 38,400 ft Weight 32,430 lb
Avg. N_2 8290 rpm Stabilizer .3 DEG. NOSE UP
Mach No. .877

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



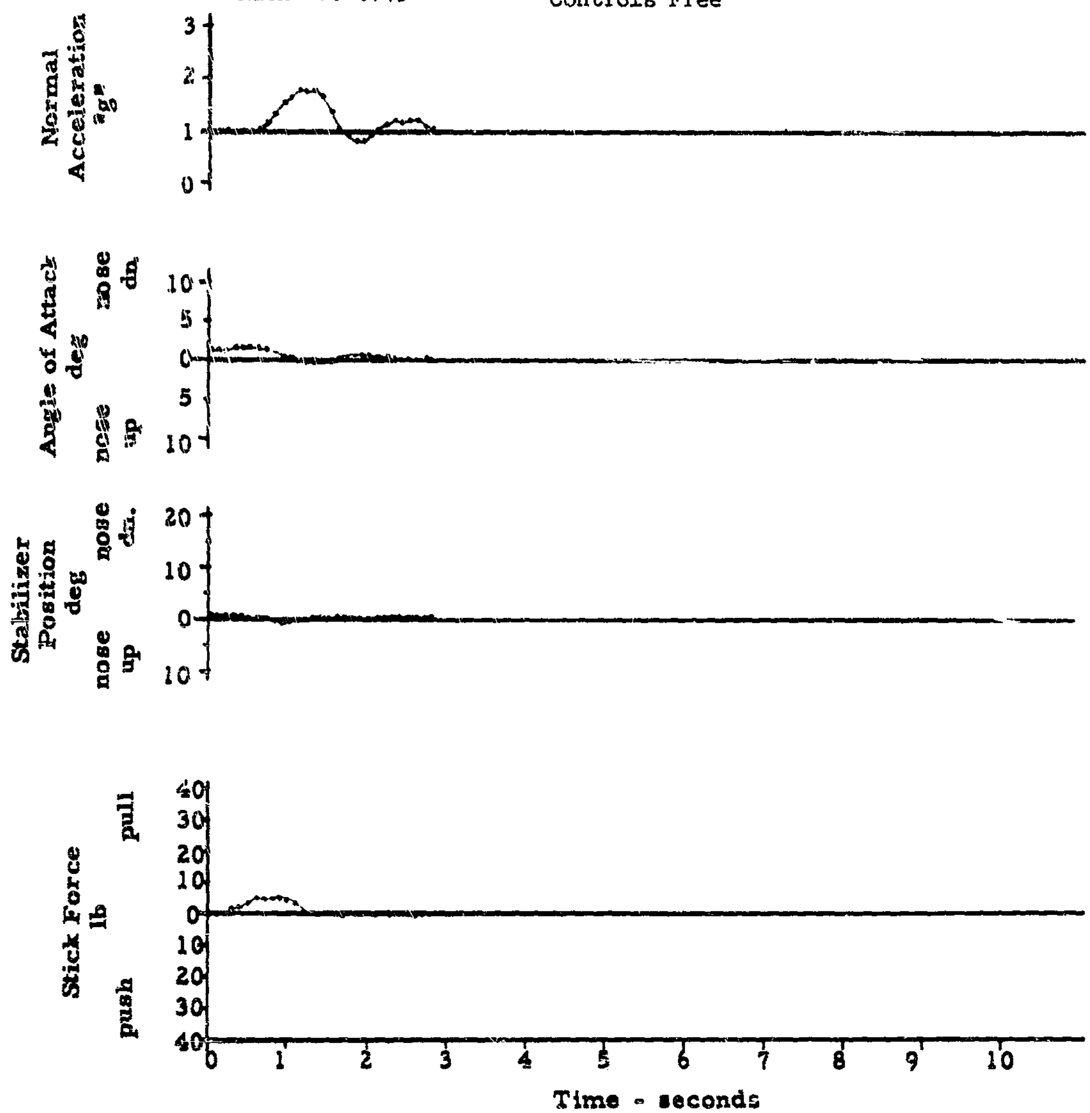
APPENDIX I

Figure No. 47

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 kts CG pos. 32.6 %MAC
Alt. 14,270 ft Weight 35,860 lb
Avg. N_2 9380 rpm Stabilizer 1.1 Deg. Nose Down
Mach No. .973 Controls Free



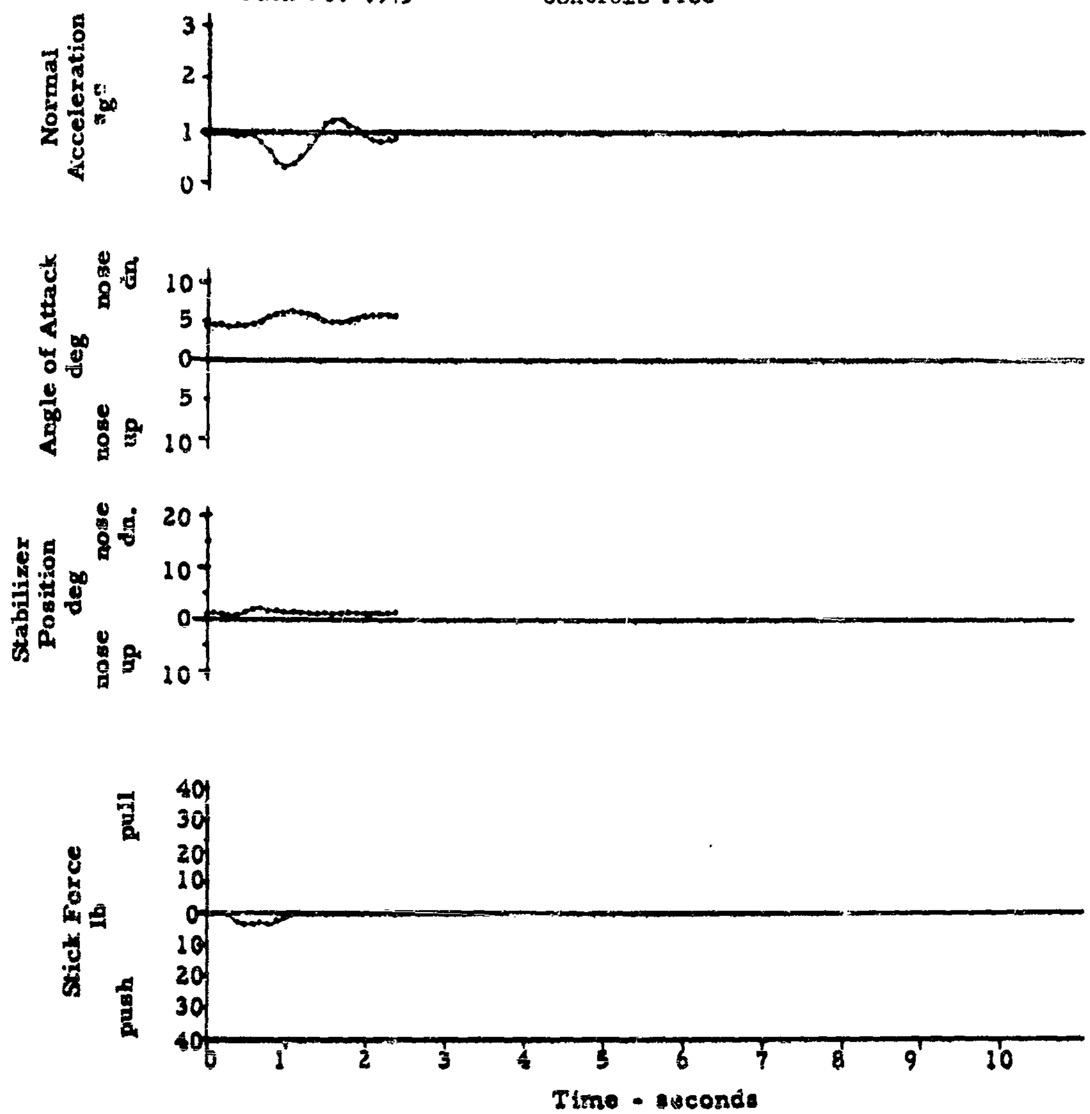
APPENDIX I

Figure No. 48

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 kts CG pos. 32.6 %MAC
Alt. 14,270 ft Weight 35,860 lb
Avg. N_2 9380 rpm Stabilizer 1.1 Deg. Nose Down
Mach No. .973 Controls Free



APPENDIX I

Figure No. 49

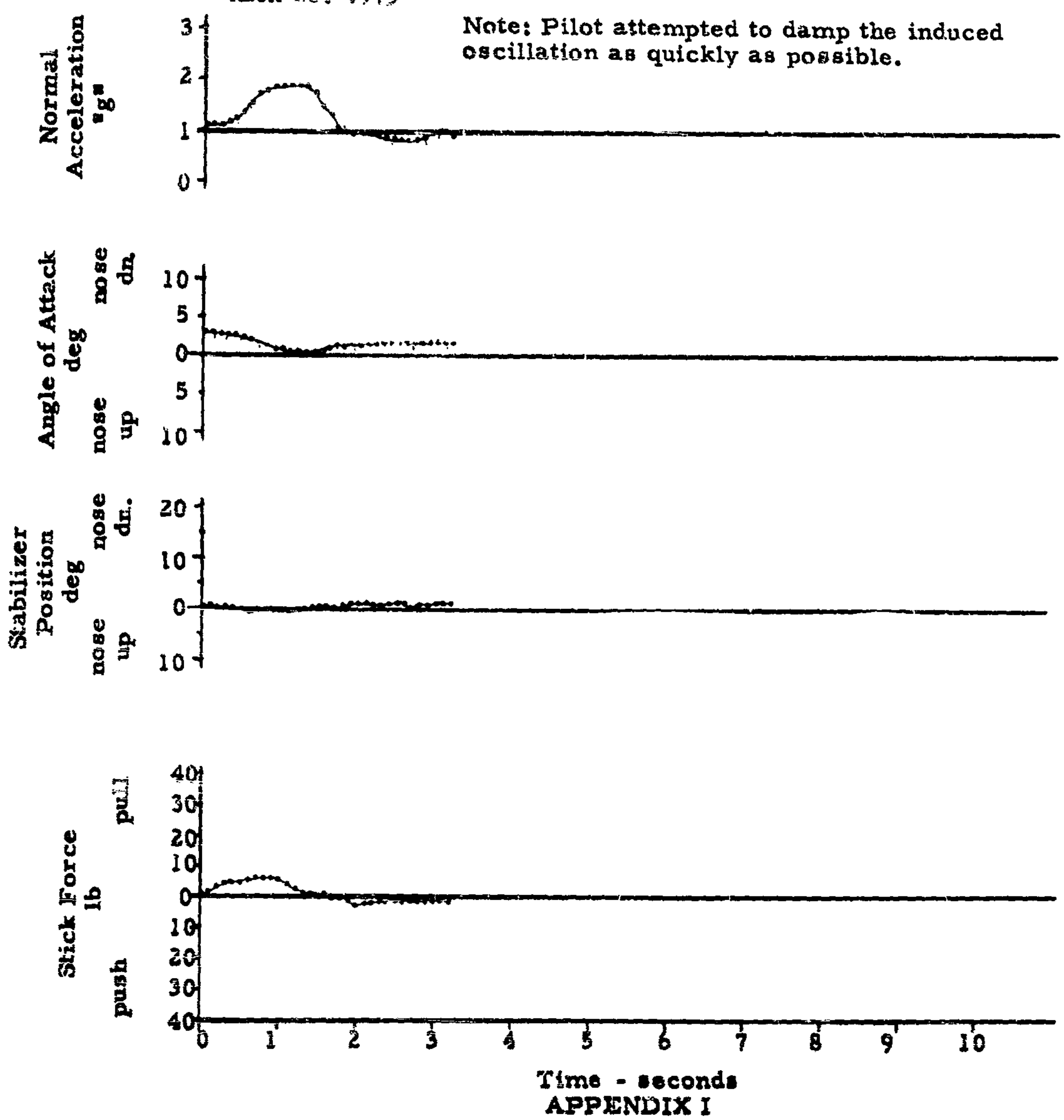
DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS	511	kts	CG pos.	32.6	%MAC
Alt.	14,270	ft	Weight	35,860	lb
Avg. N_2	9380	rpm	Stabilizer	1.1 Deg. Nose Down	

Mach No. .973

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



APPENDIX I

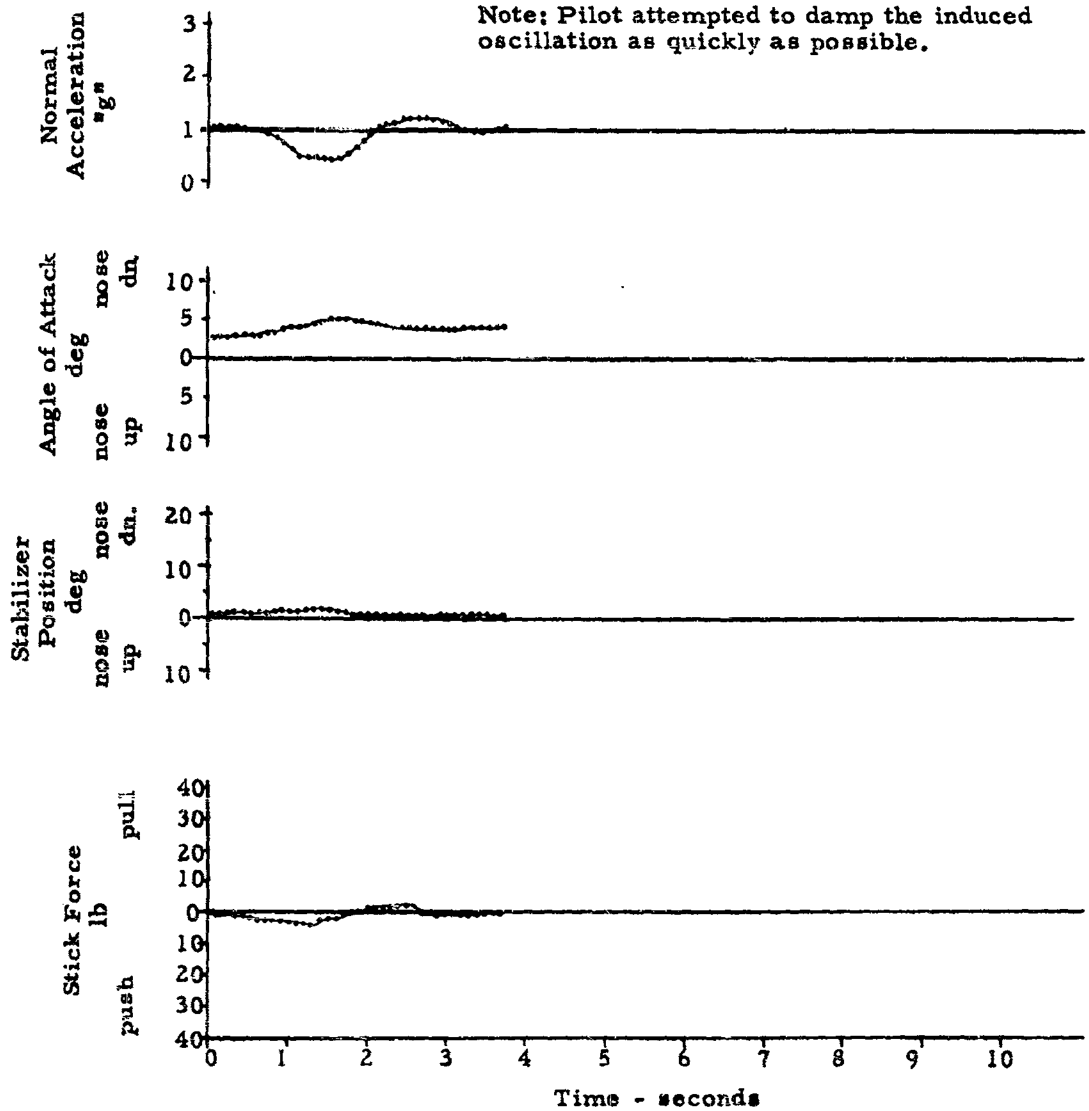
Figure No. 50

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 kts CG pos. 32.6 %MAC
Alt. 14,270 ft Weight 35,860 lb
Avg. N_2 9380 rpm Stabilizer 1.1 Deg. Nose Down
Mach No. .973

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



APPENDIX I

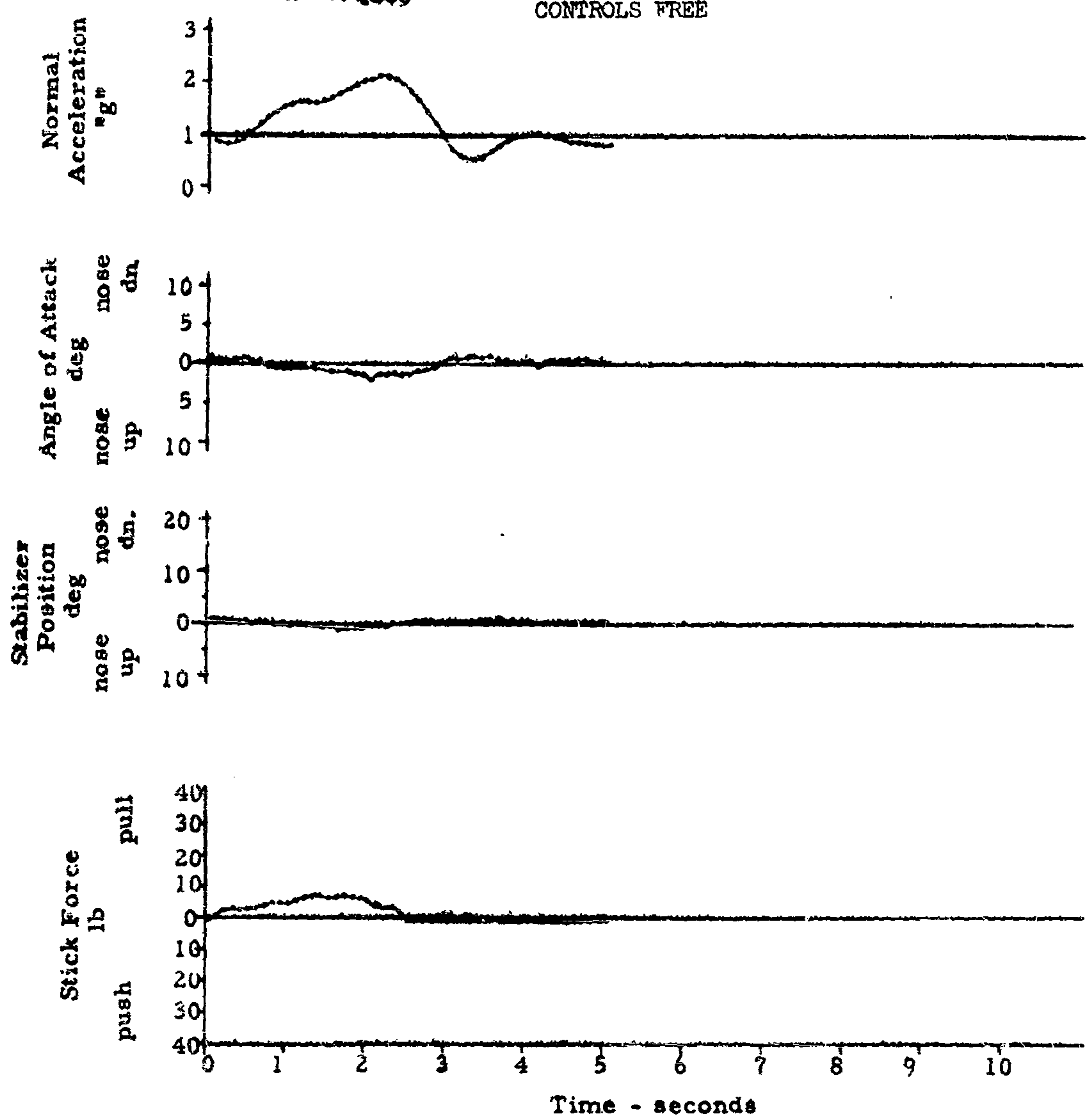
Figure No. 51

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 444 kts CG pos. 33.4 %cMAC
Alt. 10,940 ft Weight 31,870 lb
Avg. N_2 8670 rpm Stabilizer 1.1 DEG. NOSE DOWN
Mach No. .806

CONTROLS FREE



APPENDIX I

Figure No. 52

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419

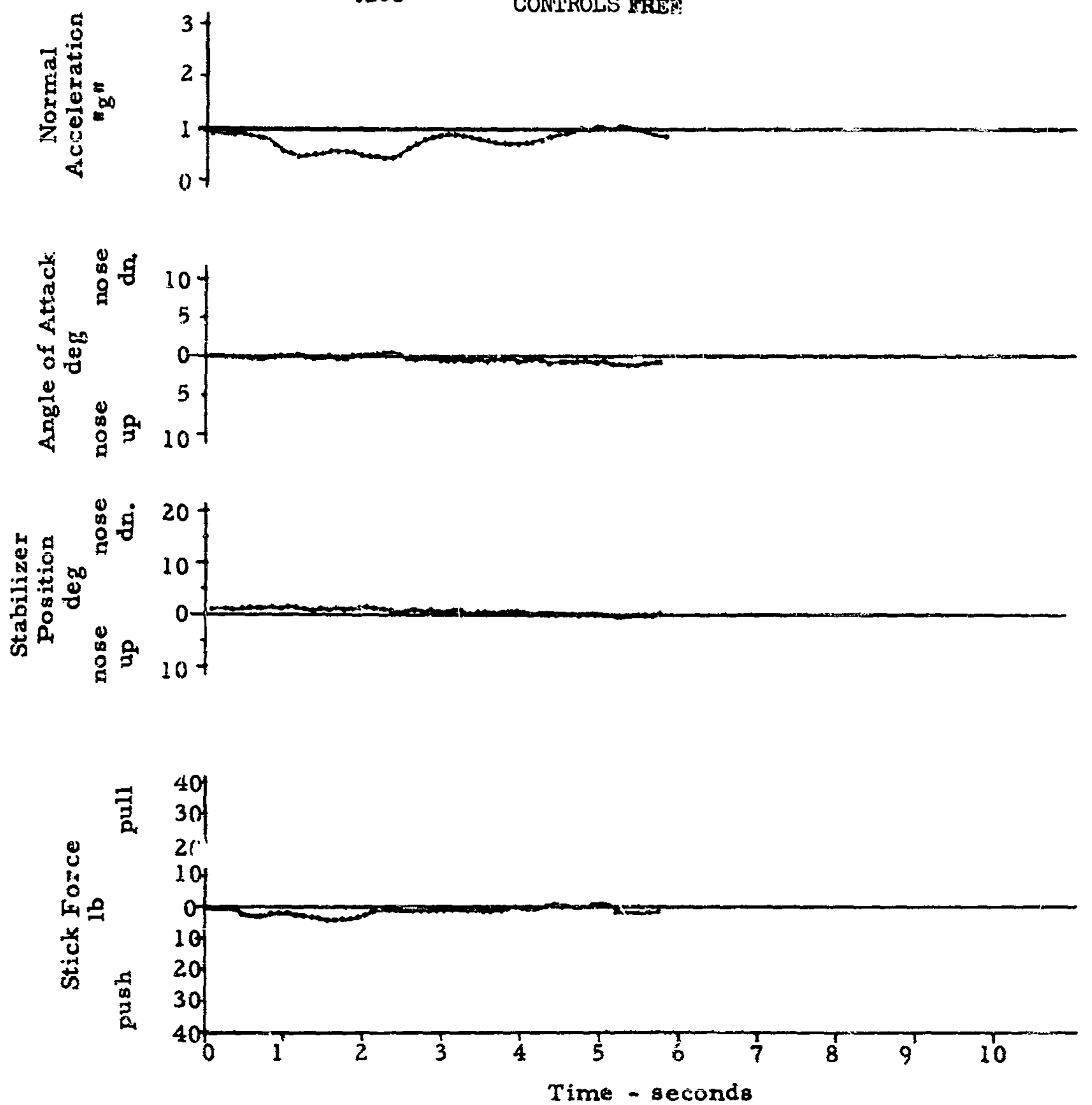
Cruise Configuration

TRIM CONDITIONS

CAS	444	kts	CG pos.	33.4	%MAC
Alt.	10,940	ft	Weight	31,870	lb
Avg. N_2	8670	rpm	Stabilizer	1.1 DEG. NOSE DOWN	

Mach No. .806

CONTROLS FREE



APPENDIX I

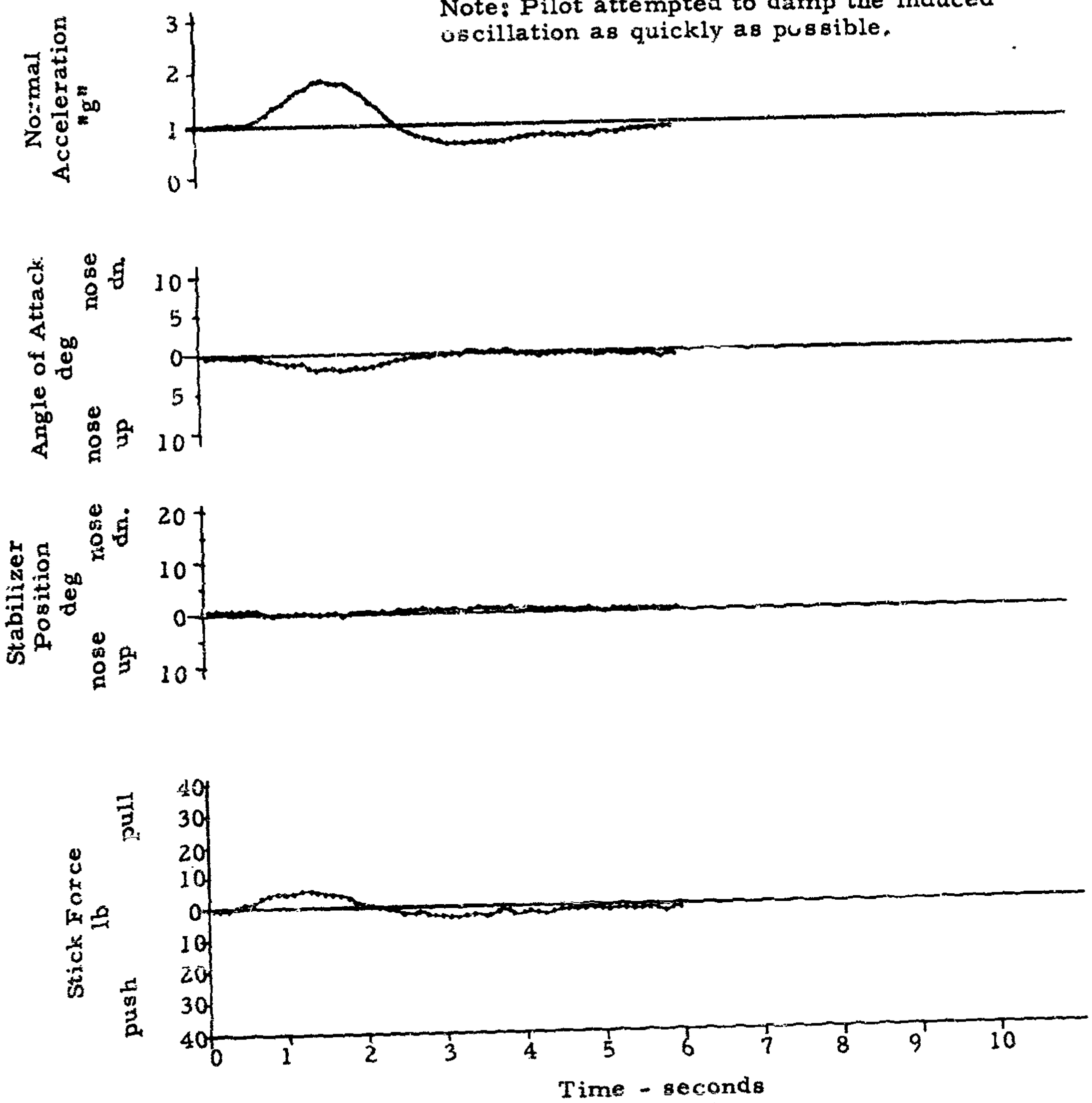
Figure No. 53

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 444 kts CG pos. 33.4 %MAC
Alt. 10,940 ft Weight 31,870 lb
Avg. N_2 8670 rpm Stabilizer 1.1 DEG. NOSE DOWN
Mach No. .806

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



APPENDIX I

Figure No. 54

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

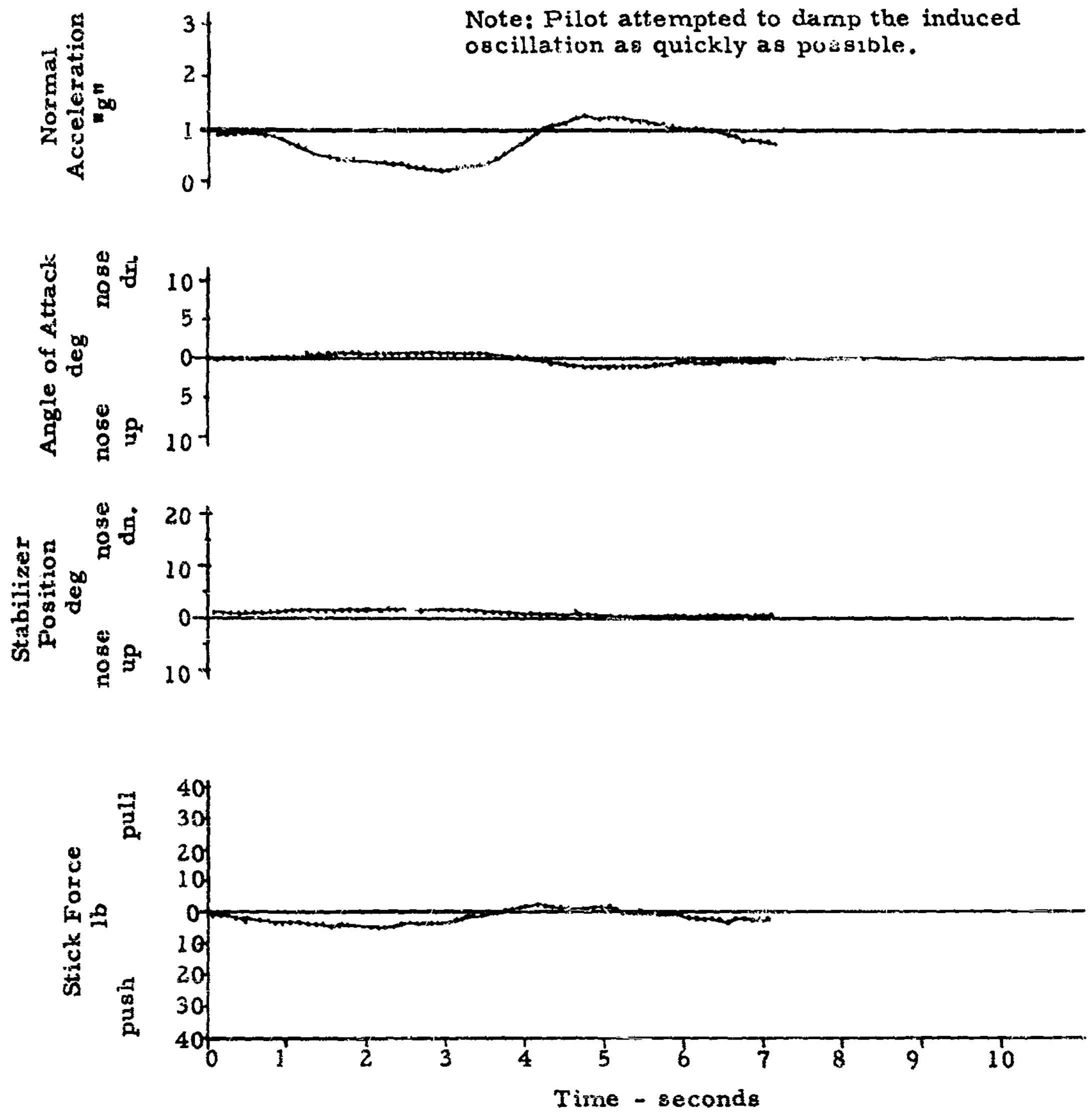
CAS 444 kts CG pos. 33.4 %MAC

Alt. 10,940 ft Weight 31,870 lb

Avg. N_2 8670 rpm Stabilizer 1.1 DEG. NOSE DOWN

Mach No. .806

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



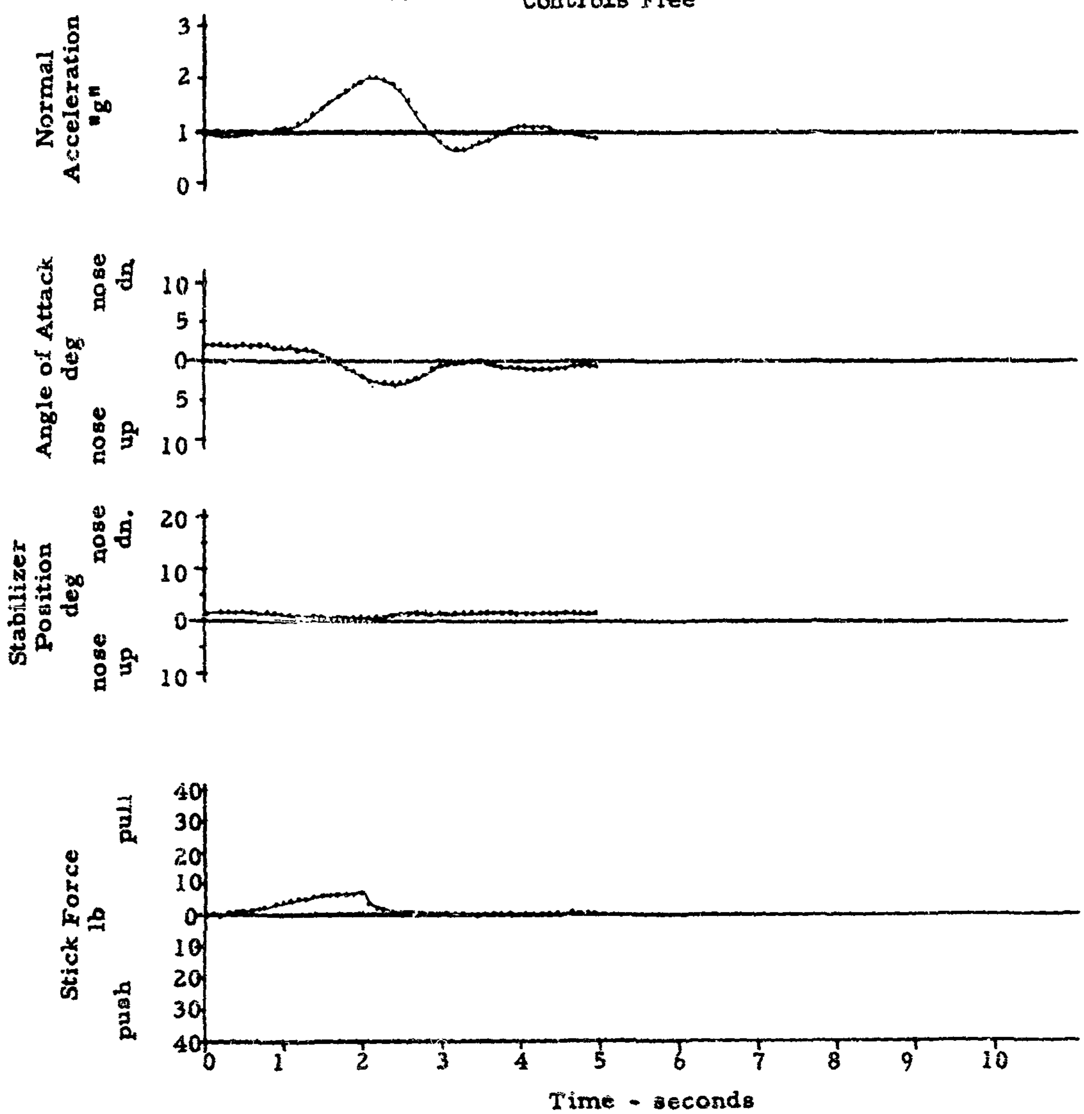
APPENDIX I

Figure No. 55

DYNAMIC LONGITUDINAL STABILITY *
Cruise Configuration
F-101A, USAF No. 53-2419

TRIM CONDITIONS

CAS 414.5 kts CG pos. 31.6 %MAC
Alt. 14,750 ft Weight 34,900 lb
Avg. N_2 8740 rpm Stabilizer 1.0 Deg. Nose Down
Mach No. .8055 Controls Free



APPENDIX I

Figure No. 56

DYNAMIC LONGITUDINAL STABILITY*

F-101A, USAF No.53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	414.5	kts	CG pos.	31.6	%MAC
Alt.	14,750	ft	Weight	34,900	lb
Avg. N_2	8740	rpm	Stabilizer	1.0 Deg. Nose Down	

Mach No. .8055 Controls Free

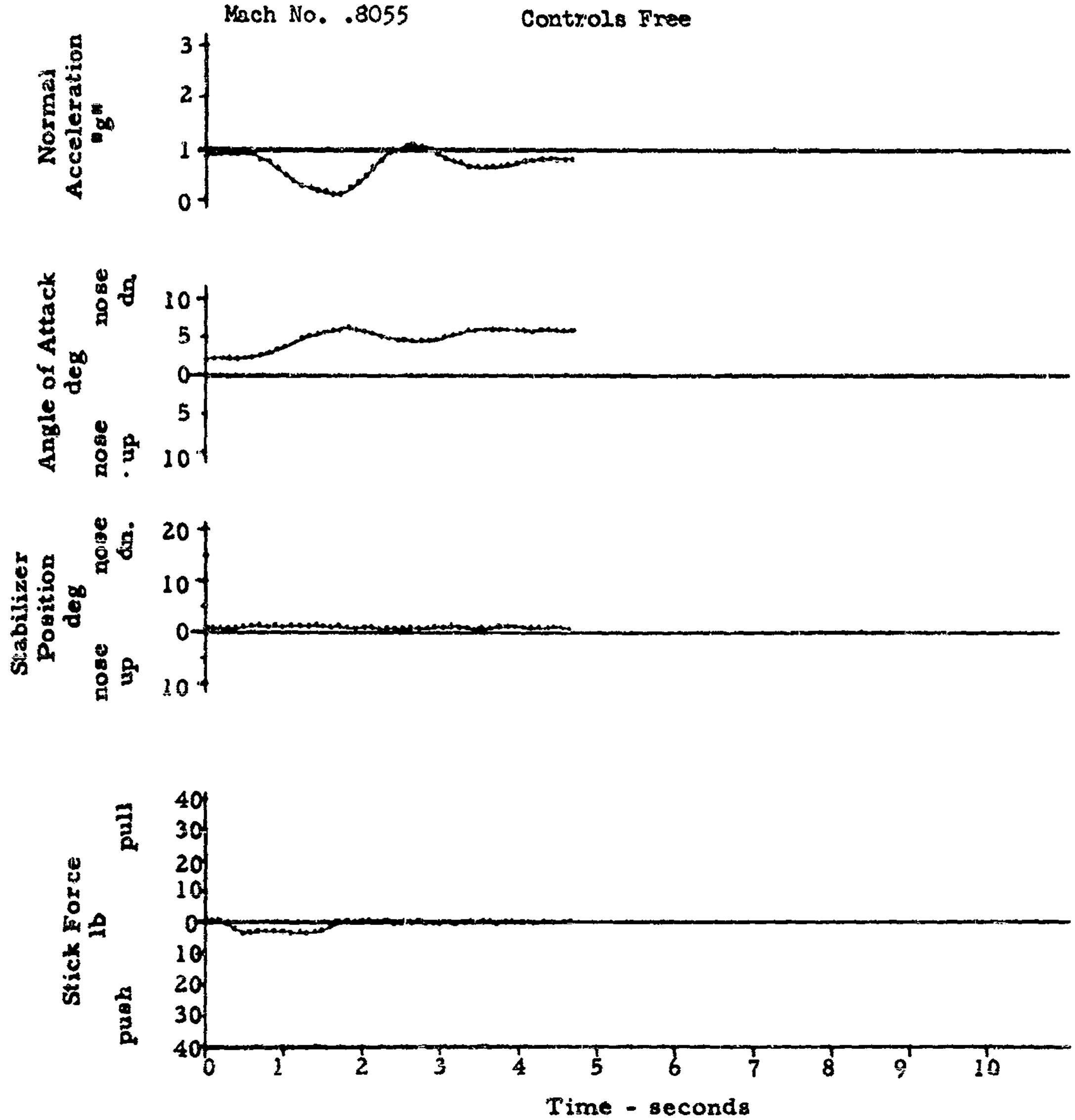


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Figure No. 57

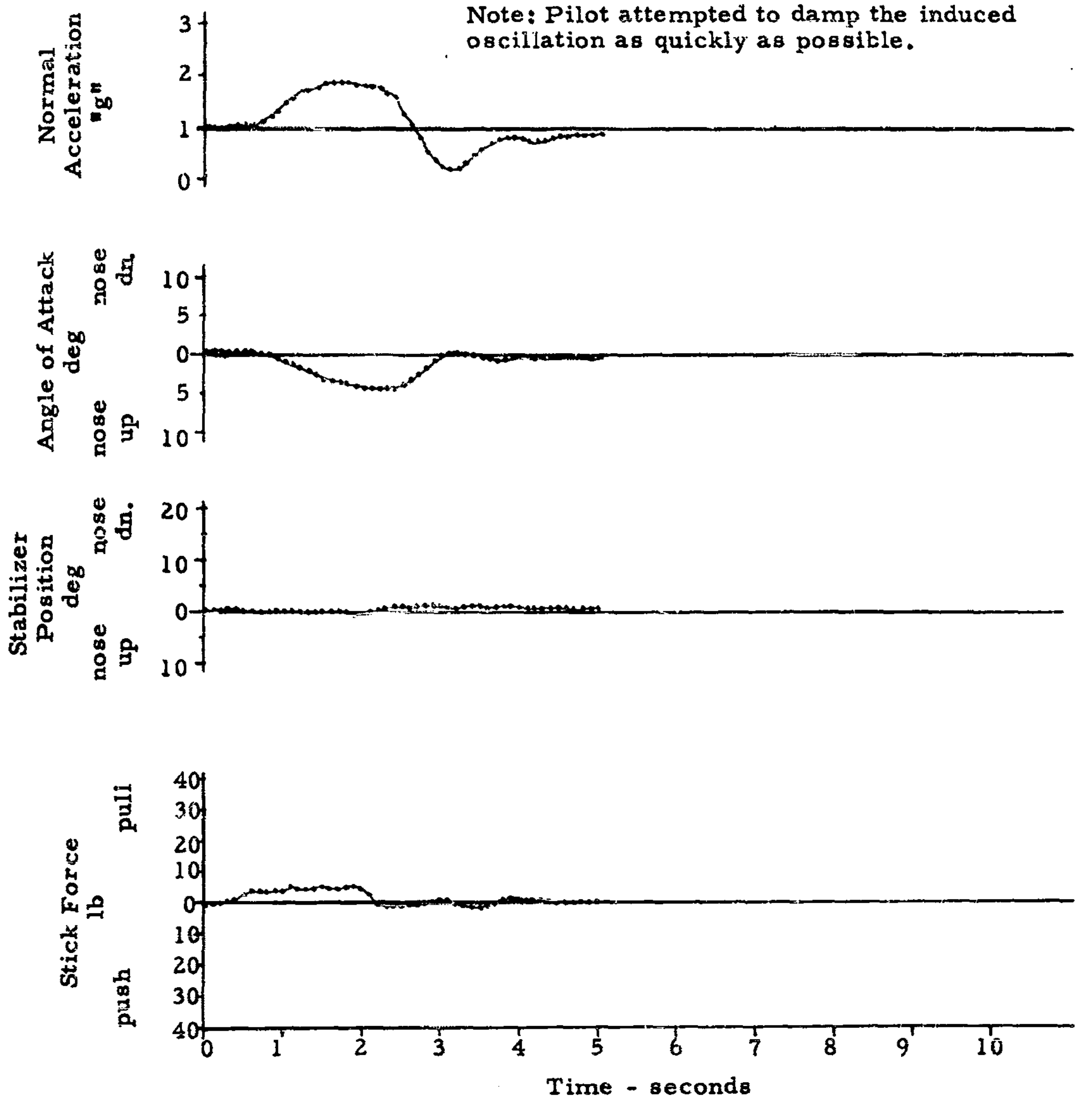
DYNAMIC LONGITUDINAL STABILITY*

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	414.5	kts	CG pos.	31.6	%MAC
Alt.	14,750	ft	Weight	34,900	lb
Avg. N_2	8740	rpm	Stabilizer	1.0 Deg. Nose Down	

Mach No. .8055

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



APPENDIX I

Figure No. 58

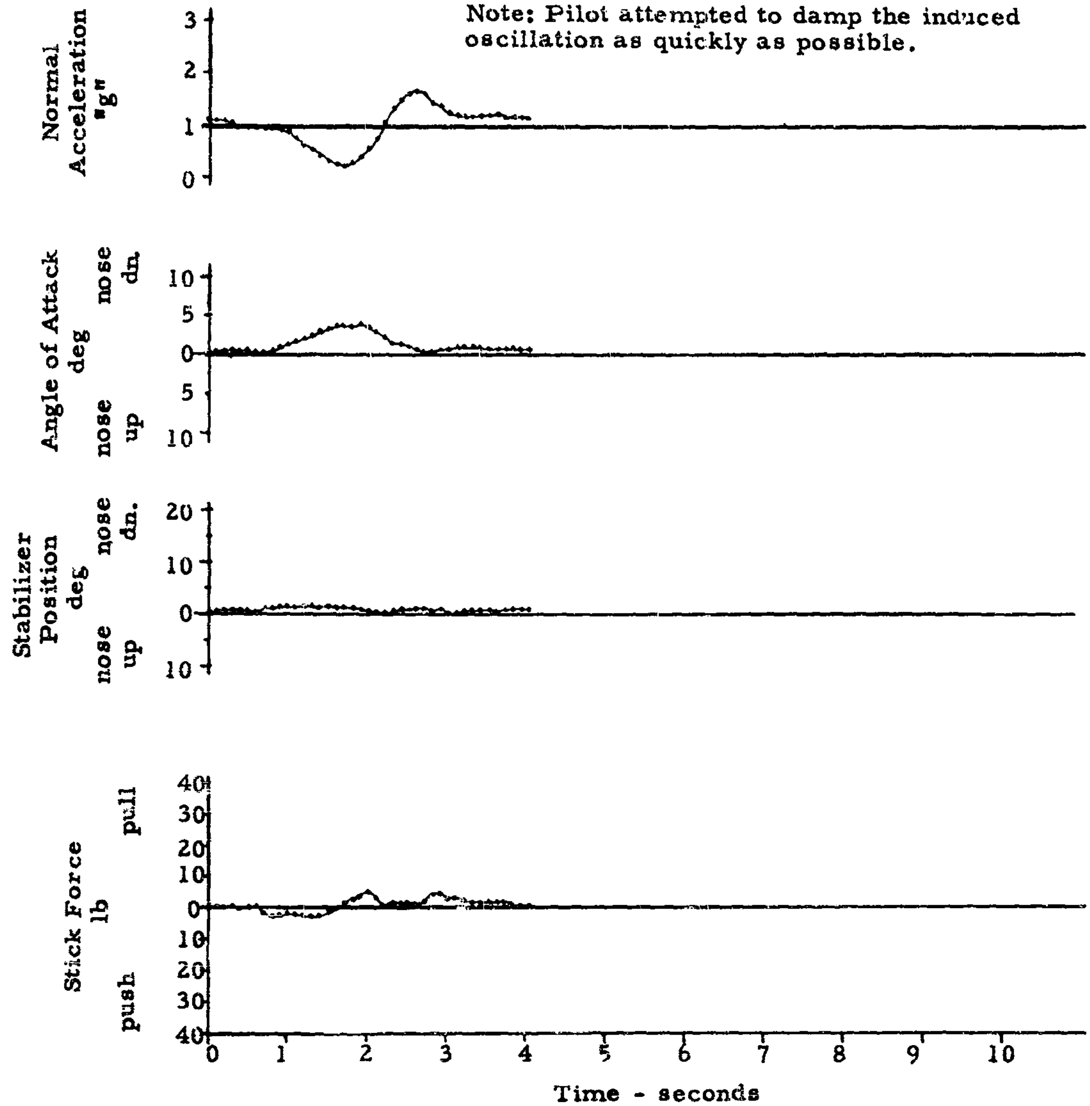
DYNAMIC LONGITUDINAL STABILITY*

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	414.5	kts	CG pos.	31.6	%MAC
Alt.	14,750	ft	Weight	34,900	lb
Avg. N_2	8740	rpm	Stabilizer	1.0 Deg. ~~Nose~~ Down	

Mach No. .8055

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



APPENDIX I

Figure No. 59

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No.53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193.5 kts CG pos. 32.1 %MAC
Alt. 12,850 ft Weight 32,120 lb
Avg. N_2 9140 rpm Stabilizer 1.5 Deg. Nose Down

Controls Free

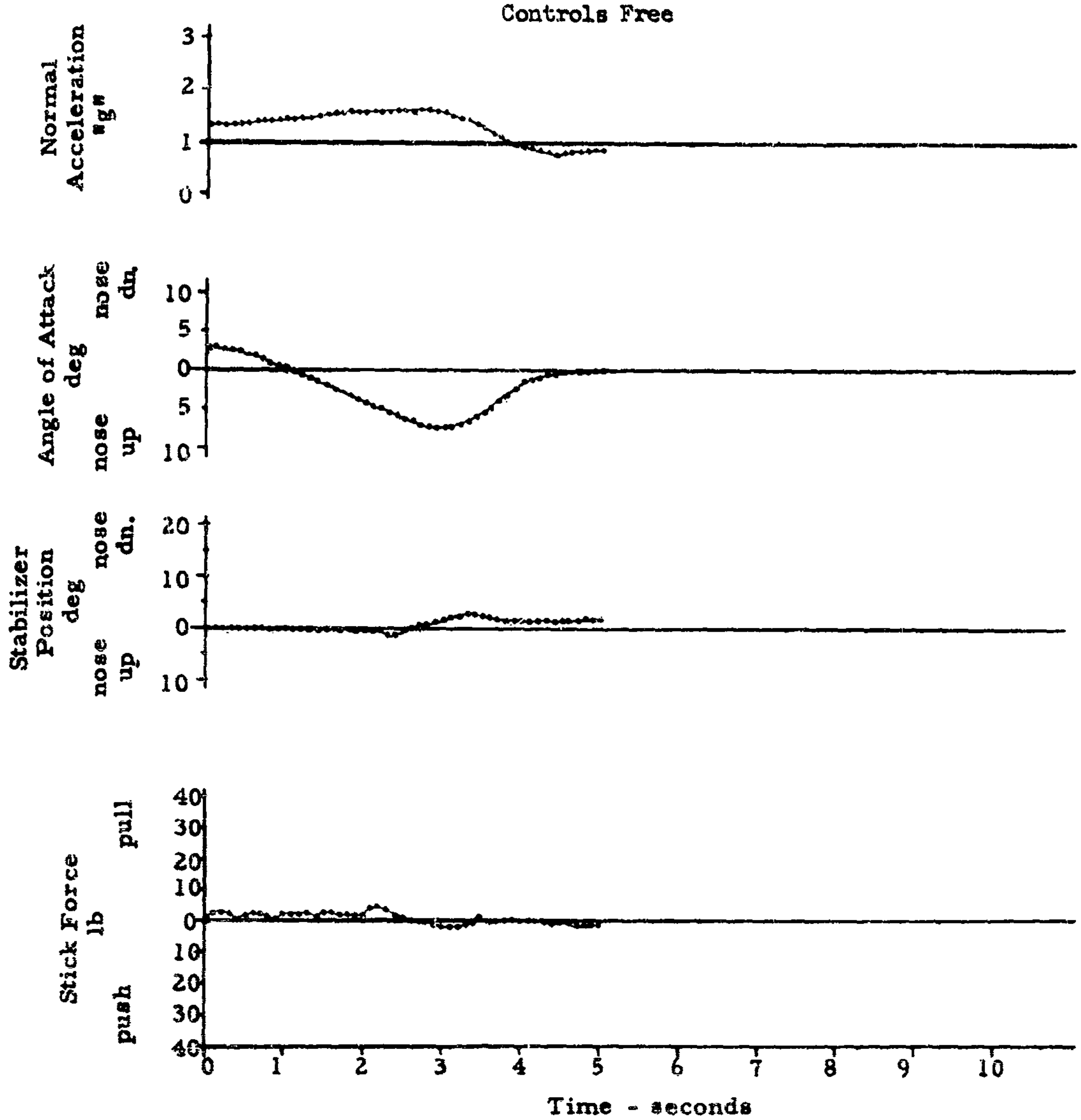


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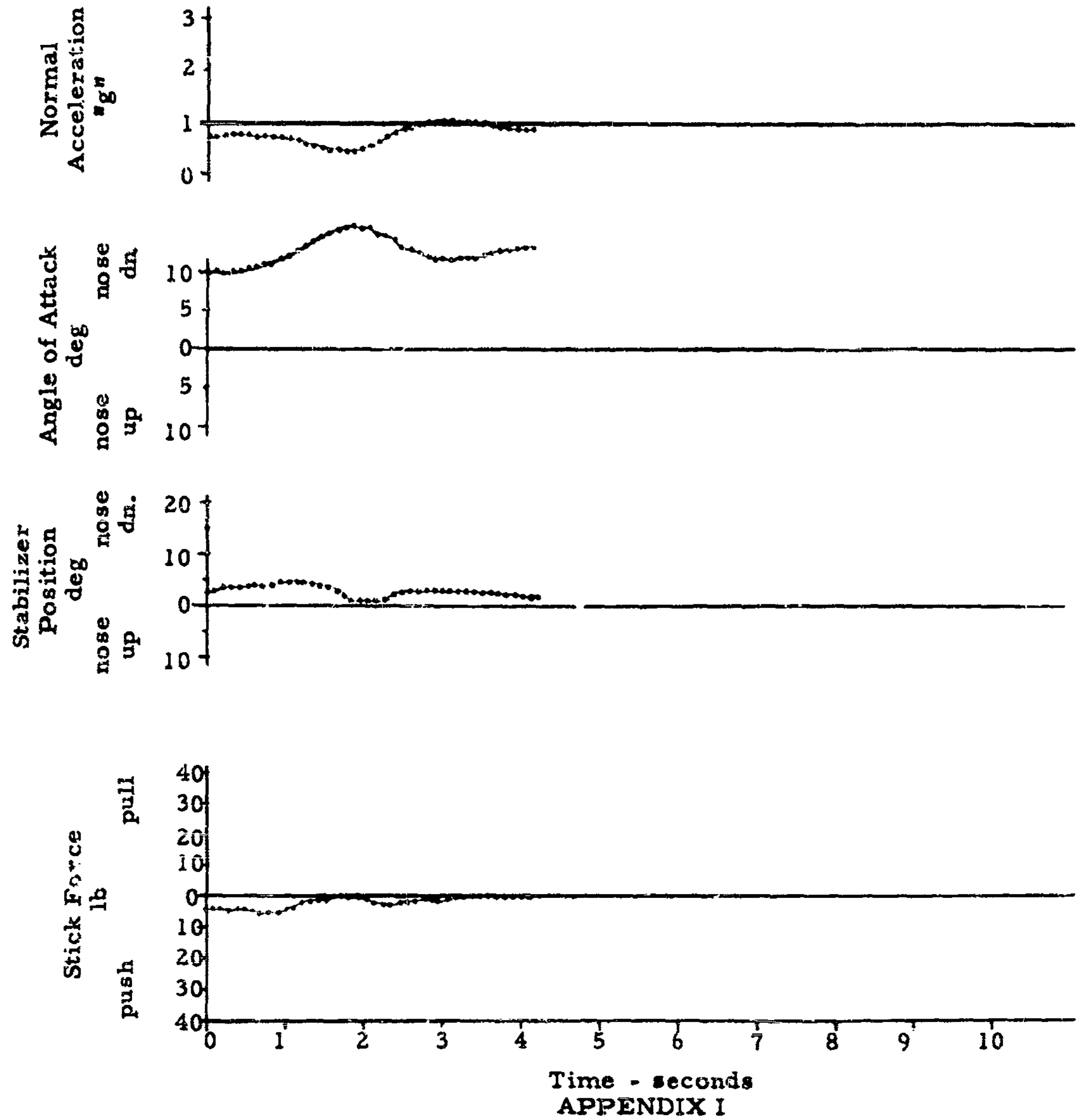
Figure No. 60

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS	193.5	kts	CG pos.	32.1	%MAC
Alt.	12,850	ft	Weight	32,120	lb
Avg. N_2	9140	rpm	Stabilizer	1.5 Deg. Nose Down	

Controls Free



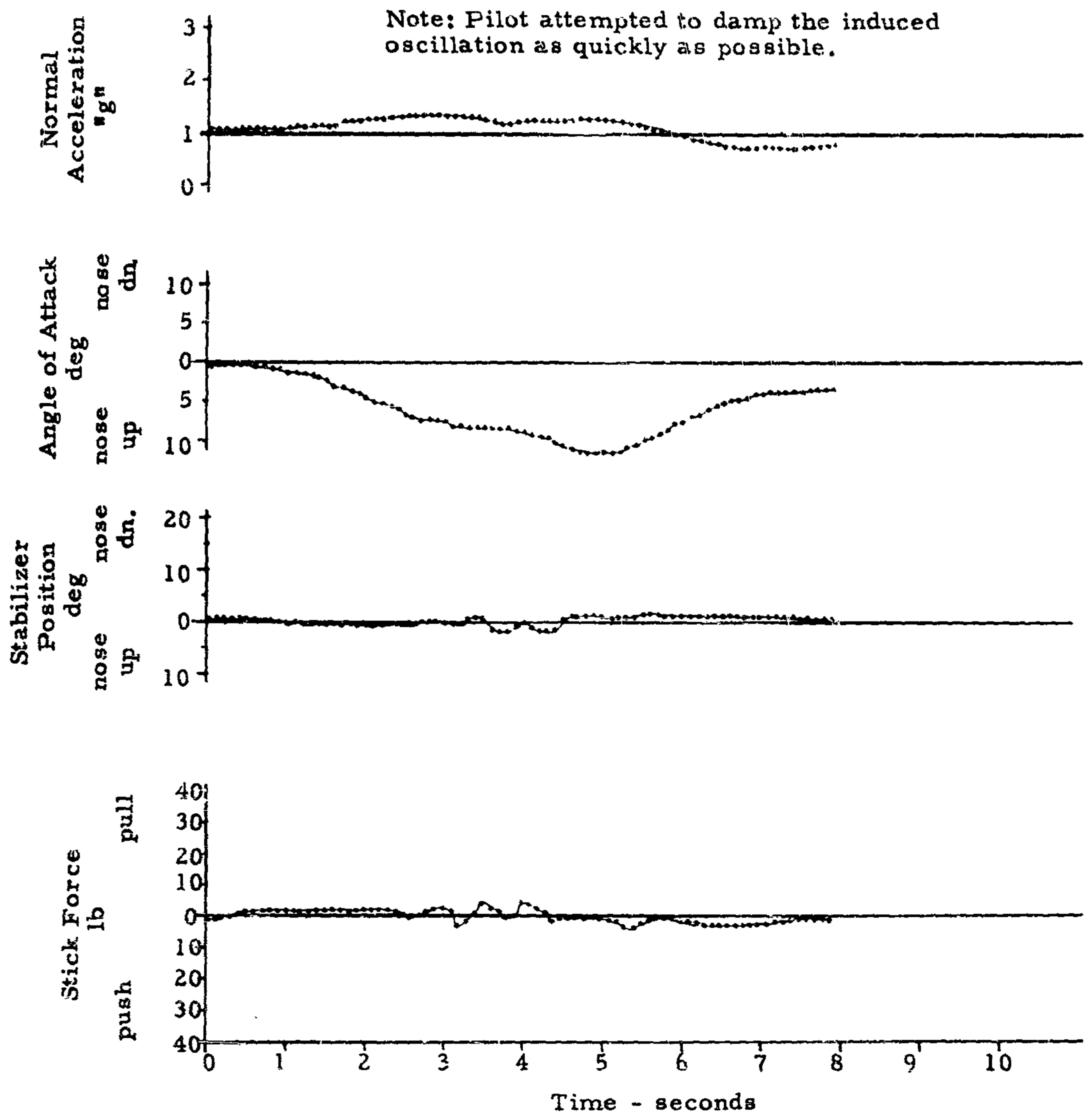
APPENDIX I

Figure No. 61

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193.5 kts CG pos. 32.1 %cMAC
Alt. 12,850 ft Weight 32,120 lb
Avg. N_2 9140 rpm Stabilizer 1.5 Deg. Nose Down



APPENDIX I

Figure No. 62

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193.5 kts CG pos. 32.1 %MAC
Alt. 12,850 ft Weight 32,120 lb
Avg. N_2 9140 rpm Stabilizer 1.5 Deg. Nose Down

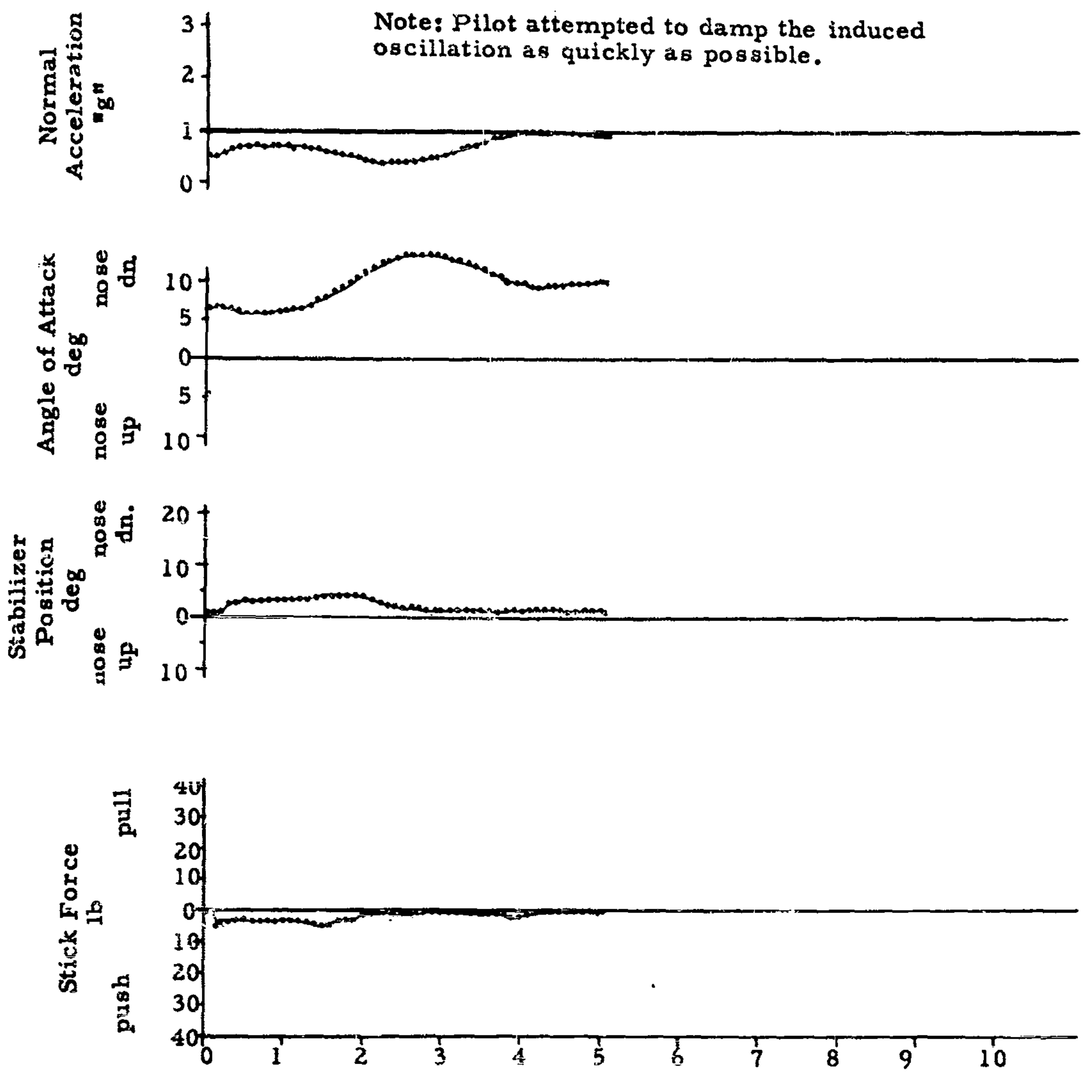


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Figure No. 63

DYNAMIC LONGITUDINAL STABILITY*

F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS	192.5	kts	CG pos.	32.2	%MAC
Alt.	16,000	ft	Weight	33,870	lb
Avg. N_2	8975	rpm	Stabilizer	0.9 Deg. Nose Down	

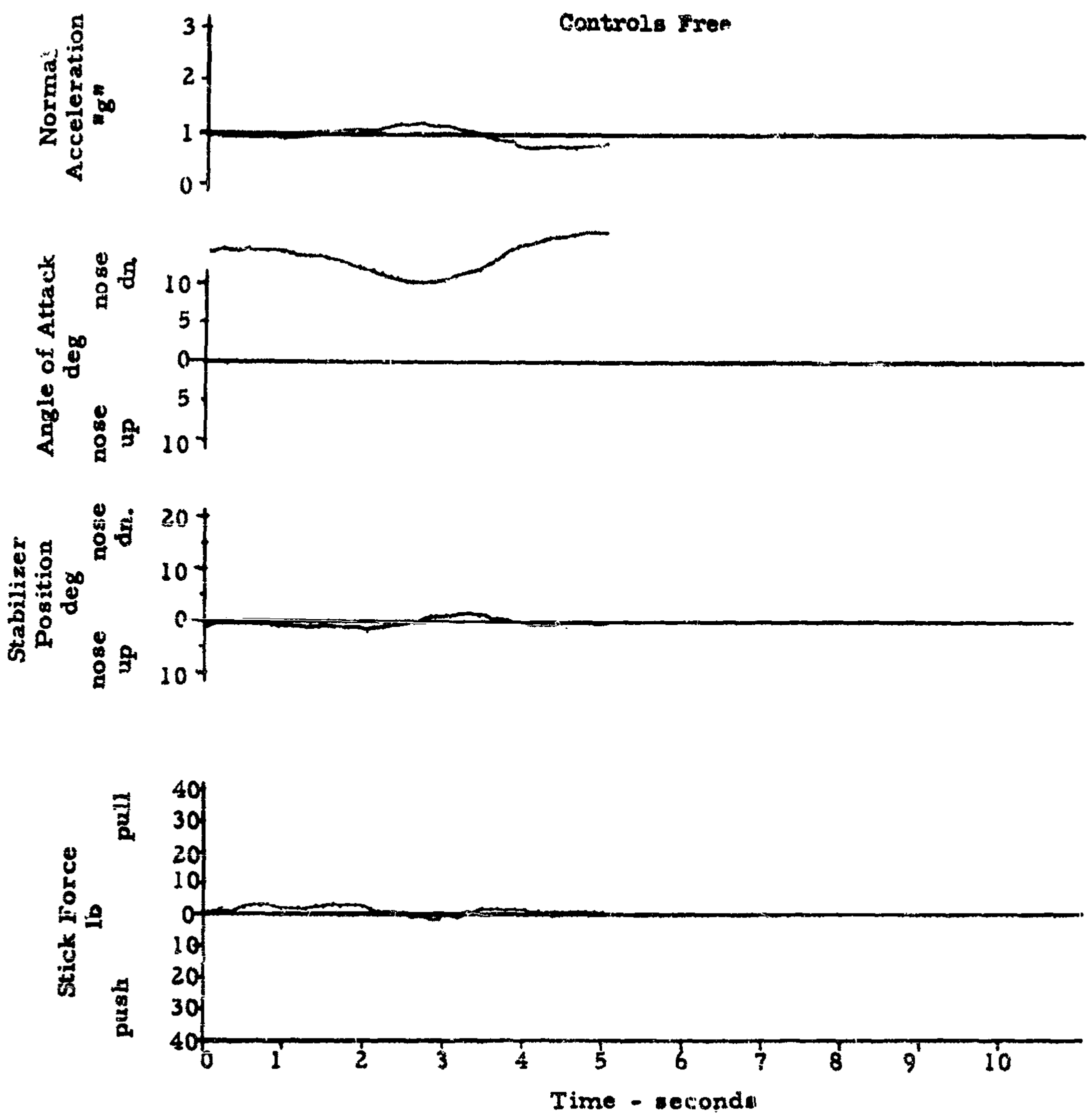

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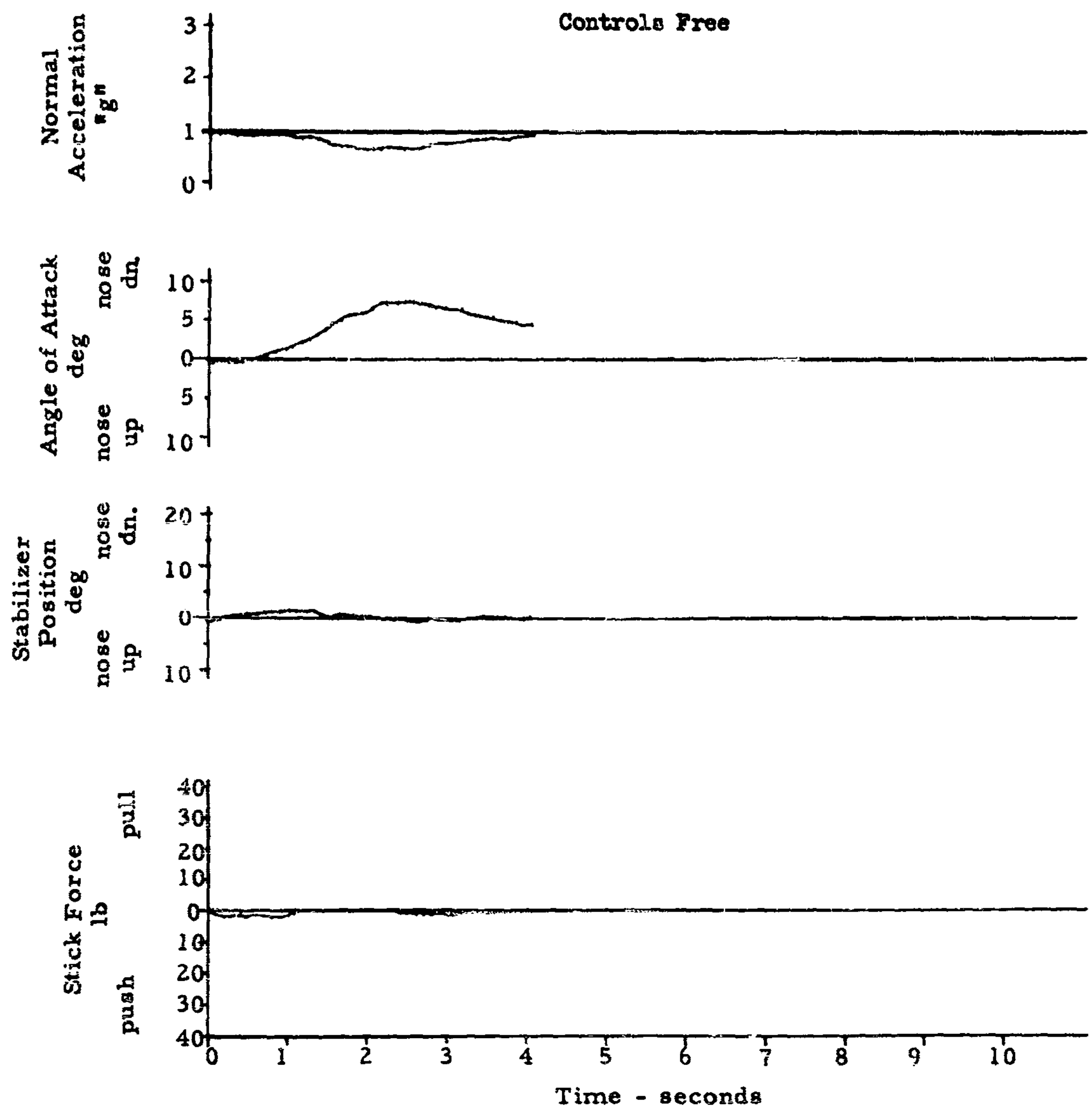
Figure No. 64

DYNAMIC LONGITUDINAL STABILITY

F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS	1925	kts	CG pos.	32.2	%MAC
Alt.	16,000	ft	Weight	33,870	lb
Avg. N_2	8975	rpm	Stabilizer	0.9 Deg. Nose Down	



APPENDIX I

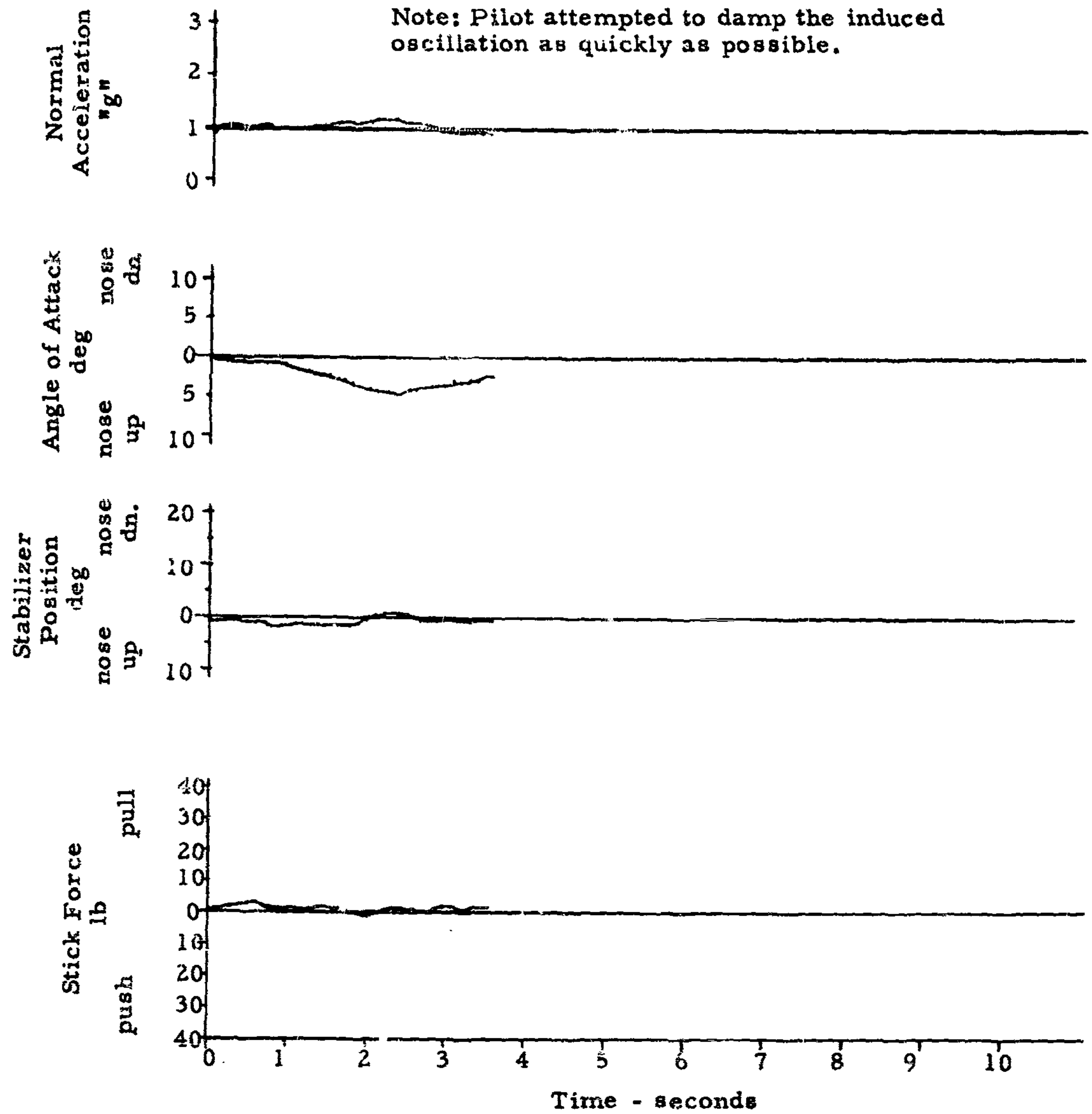
Figure No. 65

DYNAMIC LONGITUDINAL STABILITY*

F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS	192.5	kts	CG pos.	32.2	%MAC
Alt.	16,000	ft	Weight	33,870	lb
Avg. N_2	8975	rpm	Stabilizer	0.9 Deg. Nose Down	

Note: Pilot attempted to damp the induced oscillation as quickly as possible.



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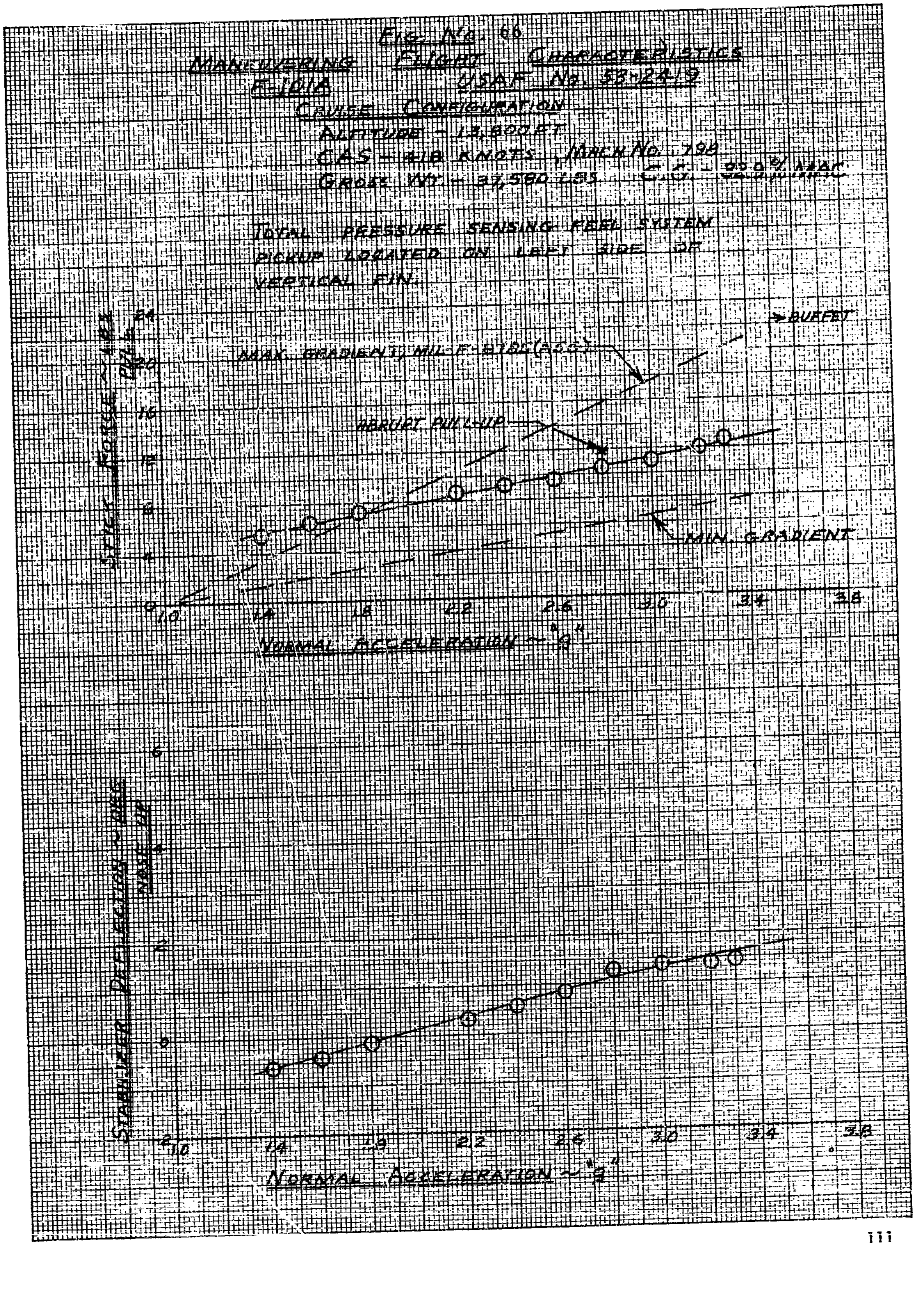


FIG. NO. 66
MANEUVERING FLIGHT CHARACTERISTICS
F-101A USAF No. 53-2419
CRUISE CONFIGURATION
ALTITUDE - 13,800 FT
CAS - 418 KNOTS, MACH NO. .798
GROSS WT. - 37,580 LBS
TOTAL PRESSURE SENSING FEEL SYSTEM
PICKUP LOCATED ON LEFT SIDE OF
VERTICAL FIN.
BUFFET
MAX. GRADIENT
ABRUPT PULL-UP
MIN. GRADIENT
STICK FORCE ~ LBS
PULL
NORMAL ACCELERATION ~ "g"
STABILIZER DEFLECTION ~ DEG
NOSE UP
NORMAL ACCELERATION ~ "g"

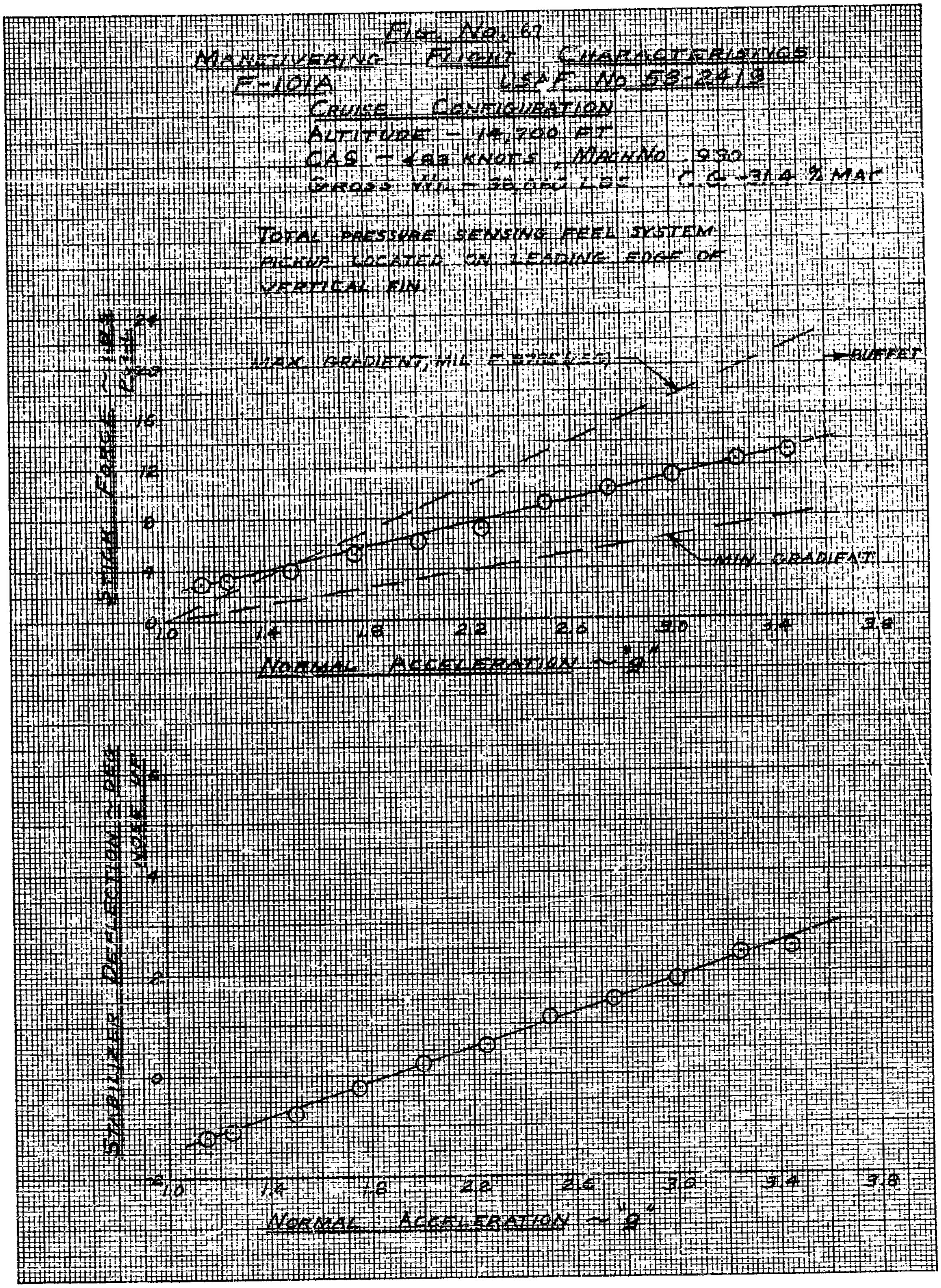


FIG. NO. 61
MANEUVERING FLIGHT CHARACTERISTICS
F-101A USAF No 53-2419
CRUISE CONFIGURATION
ALTITUDE – 14,700 FT
CAS – 483 KNOTS, MACH NO .930
C.G. – 31.4 % MAC
TOTAL PRESSURE SENSING FEEL SYSTEM
PICKUP LOCATED ON LEADING EDGE OF
VERTICAL FIN
MAX GRADIENT, MIL
BUFFET
MIN GRADIENT
STICK FORCE ~ LBS
0
4
8
12
16
24
1.0
1.4
1.8
2.2
2.6
3.0
3.4
3.8
NORMAL ACCELERATION ~ "g"
STABILIZER DEFLECTION ~ DEG
6
4
2
0
-2
1.0
1.4
1.8
2.2
2.6
3.0
3.4
3.8
NORMAL ACCELERATION ~ "g"

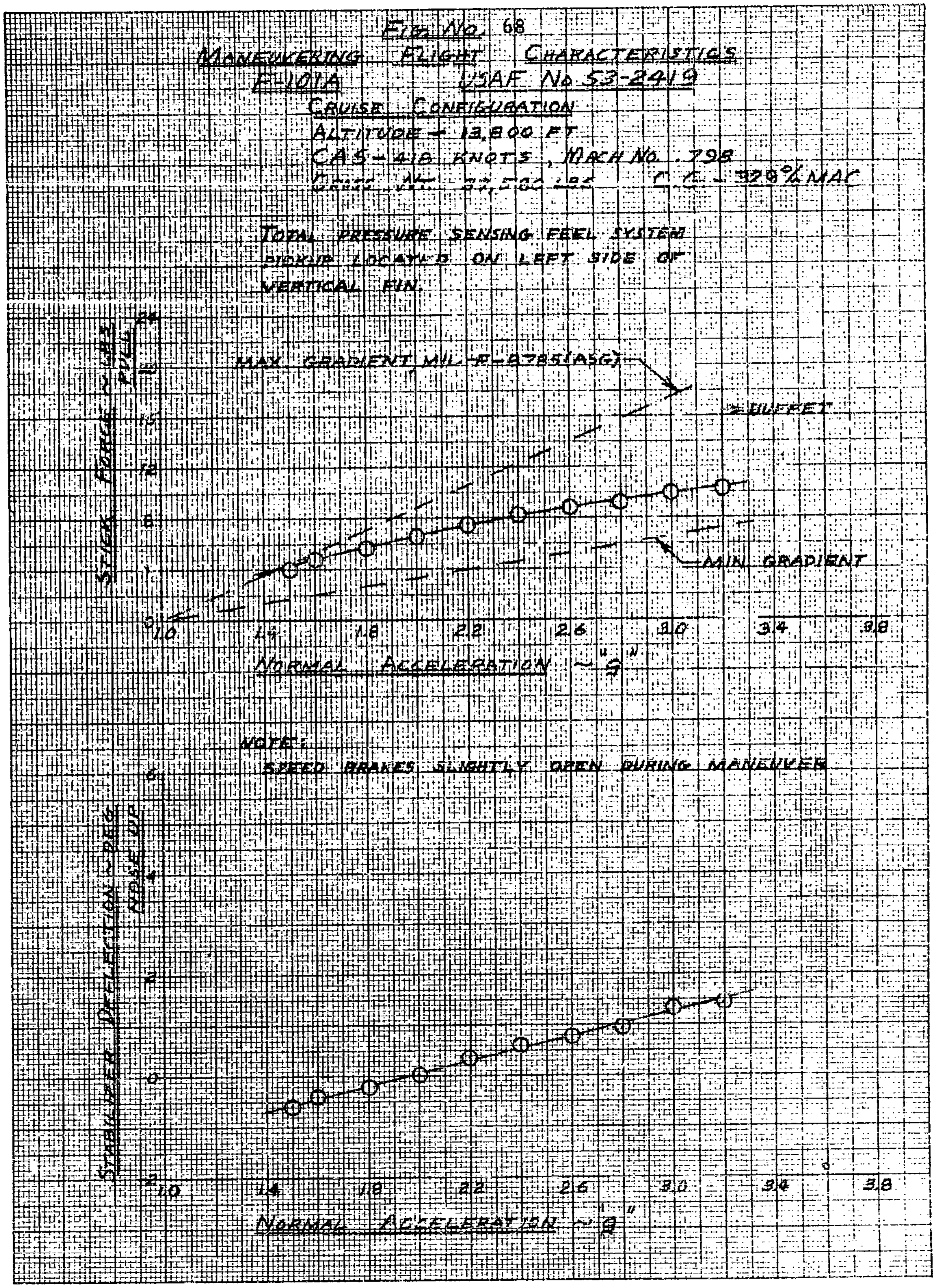


FIG. NO. 68
MANEUVERING FLIGHT CHARACTERISTICS
F-101A USAF No. 53-2419
CRUISE CONFIGURATION
ALTITUDE – 13,800 FT
CAS – 418 KNOTS, MACH No. .798
C.G. – 32.9% MAC
TOTAL PRESSURE SENSING FEEL SYSTEM
PICKUP LOCATED ON LEFT SIDE OF
VERTICAL FIN.
STICK FORCE ~ LBS PULL
MAX. GRADIENT MIL-F-8785(ASG)
BUFFET
MIN. GRADIENT
NORMAL ACCELERATION ~ "g"
NOTE:
SPEED BRAKES SLIGHTLY OPEN DURING MANEUVER
STABILIZER DEFLECTION ~ DEG NOSE UP
NORMAL ACCELERATION ~ "g"

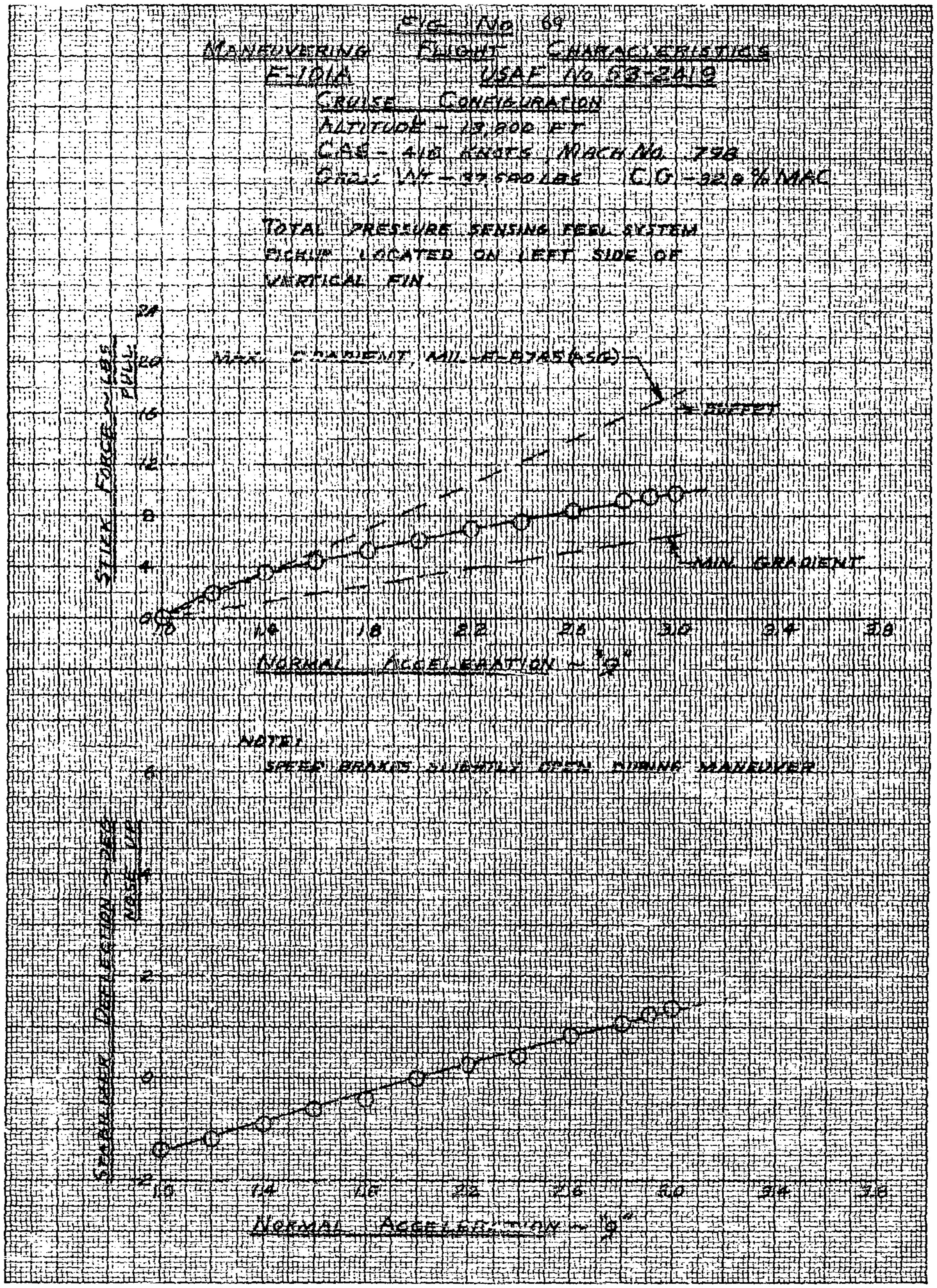


FIG. No 69
MANEUVERING FLIGHT CHARACTERISTICS
F-101A USAF No 53-2419
CRUISE CONFIGURATION
ALTITUDE - 13,800 FT
CAS - 410 KNOTS MACH No .798
TOTAL PRESSURE SENSING FEEL SYSTEM
PICKUP LOCATED ON LEFT SIDE OF
VERTICAL FIN.
STICK FORCE ~ LBS PULL
BUFFET
MIN. GRADIENT
NORMAL ACCELERATION ~ "g"
NOTE:
SPEED BRAKES PARTIALLY OPEN DURING MANEUVER
STABILIZER DEFLECTION ~ DEG NOSE UP
NORMAL ACCELERATION ~ "g"

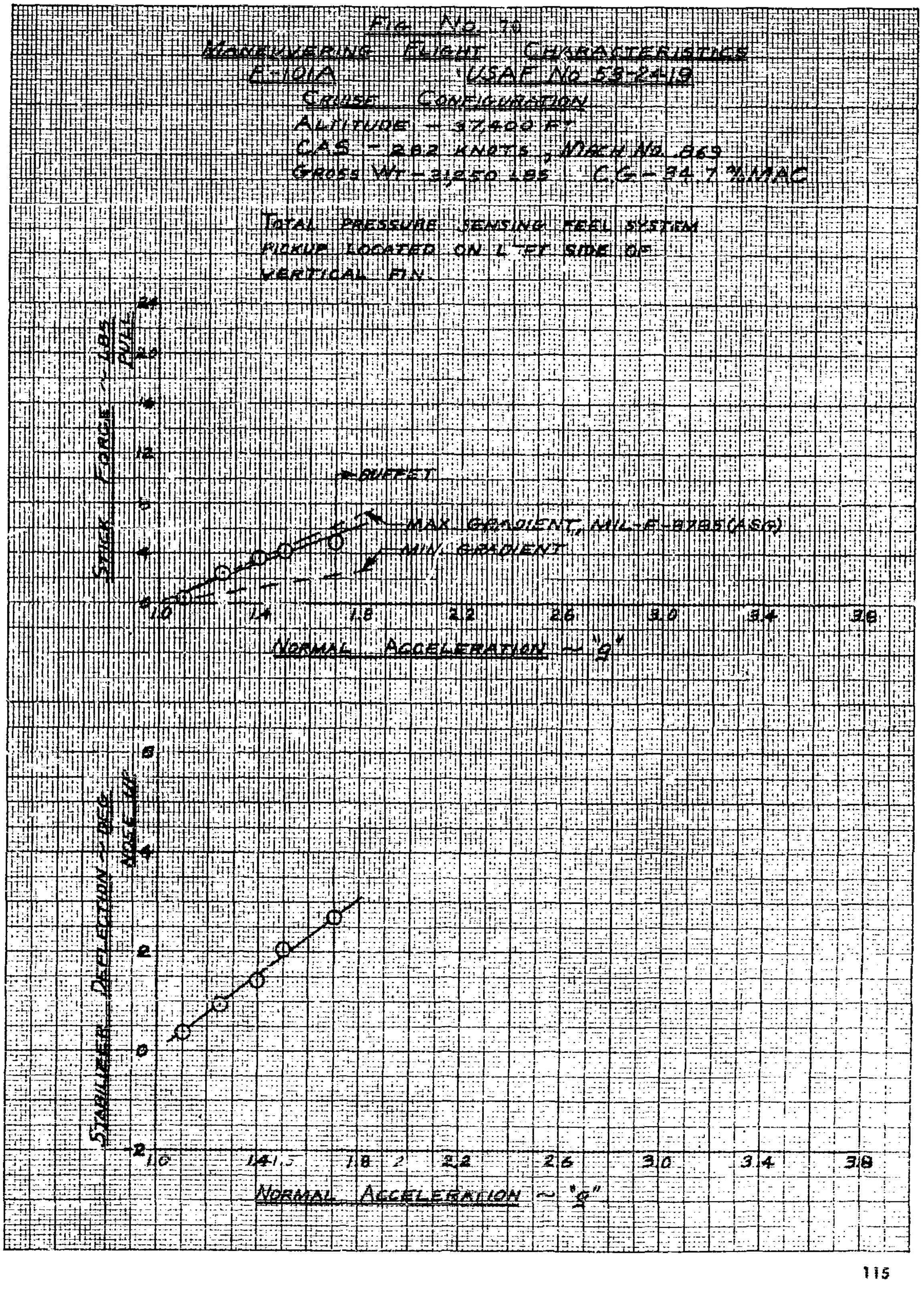


FIG. NO.
MANEUVERING FLIGHT CHARACTERISTICS
F-101A USAF NO. 53-2419
CRUISE CONFIGURATION
ALTITUDE - 37,400 FT
CAS - 282 KNOTS, MACH NO. .869
GROSS WT - 31,250 LBS C.G. - 34.7 % MAC
TOTAL PRESSURE SENSING FEEL SYSTEM
PICKUP LOCATED ON L FT SIDE OF
VERTICAL FIN
STICK FORCE ~ LBS PULL
BUFFET
MAX GRADIENT, MIL-F-8785 (ASG)
MIN. GRADIENT
0
4
8
12
16
20
24
1.0
1.4
1.8
2.2
2.6
3.0
3.4
3.8
NORMAL ACCELERATION ~ "g"
STABILIZER DEFLECTION ~ DEG NOSE UP
-2
0
2
4
6
1.0
1.4
1.5
1.8
2
2.2
2.6
3.0
3.4
3.8
NORMAL ACCELERATION ~ "g"

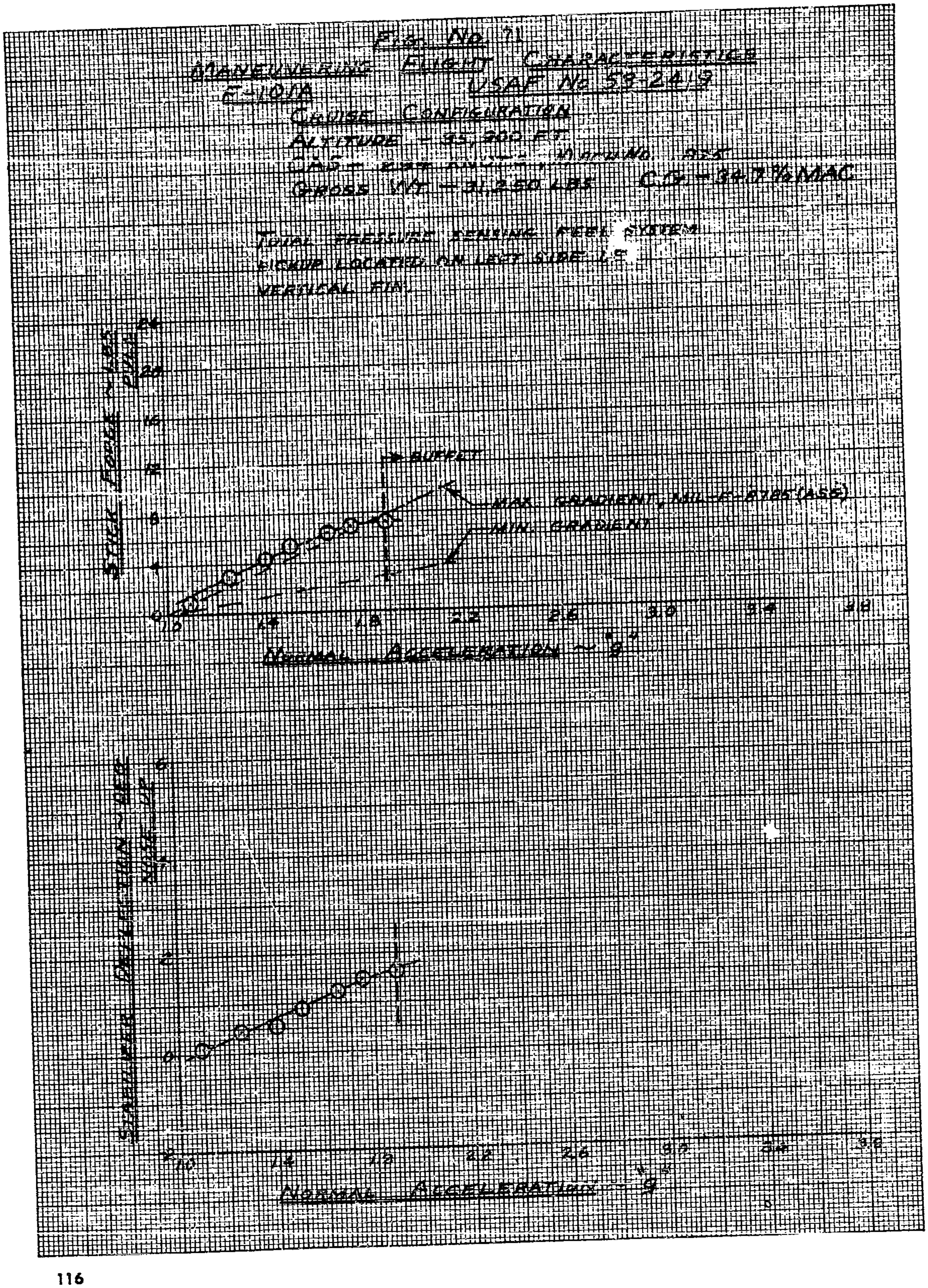


FIG. NO. 71
MANEUVERING FLIGHT CHARACTERISTICS
F-101A
USAF No 53-2419
CRUISE CONFIGURATION
GROSS WT — 31,250 LBS
C.G. — 34.7 % MAC
TOTAL PRESSURE SENSING FEEL SYSTEM
PICKUP LOCATED ON LEFT SIDE OF
VERTICAL FIN.
STICK FORCE ~ LBS PULL
BUFFET
MAX. GRADIENT, MIL-F-8785 (ASG)
MIN. GRADIENT
NORMAL ACCELERATION ~ g
STABILIZER DEFLECTION ~ DEG NOSE UP
NORMAL ACCELERATION ~ g

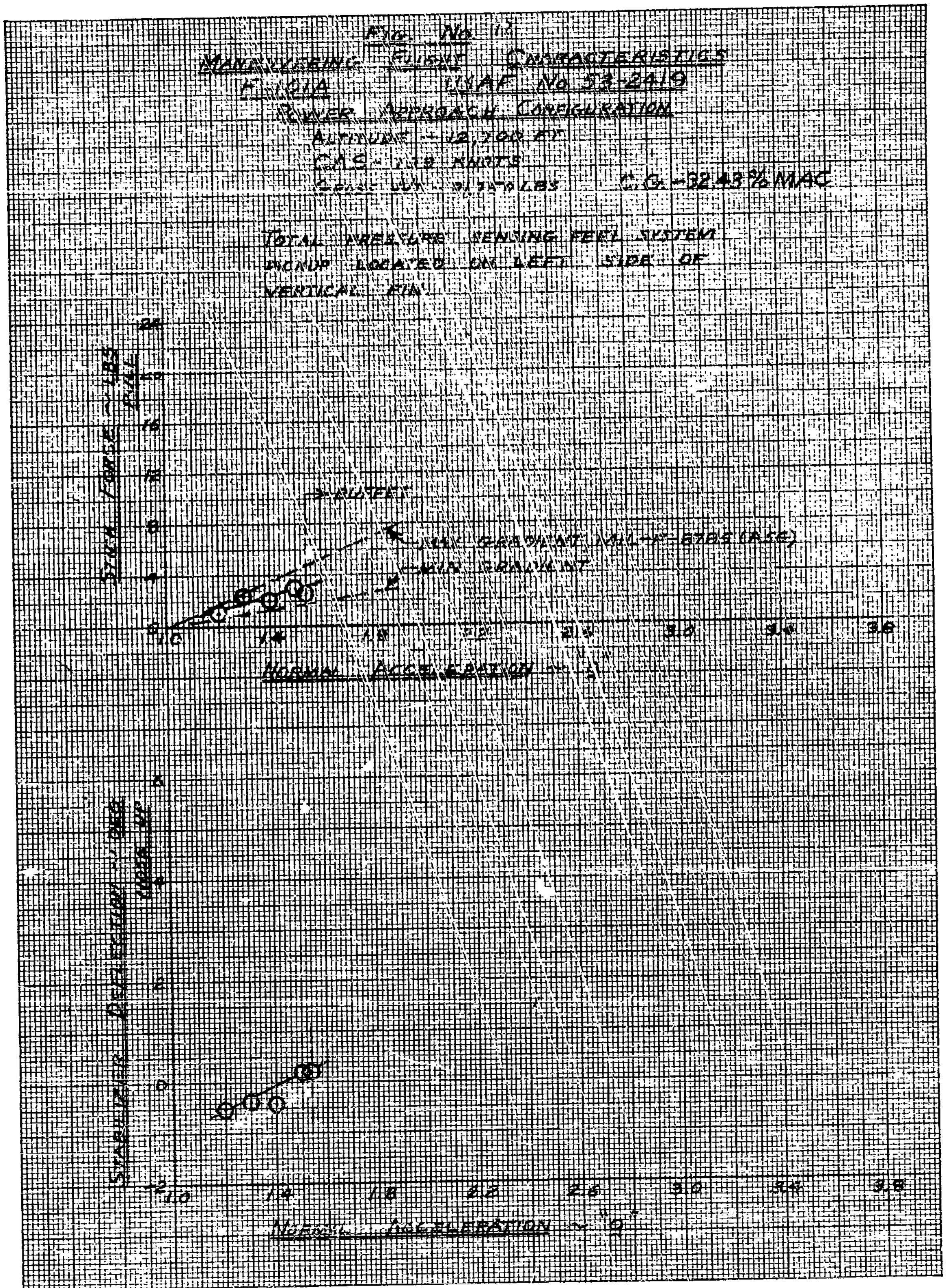

MANEUVERING FLIGHT CHARACTERISTICS
USAF No 53-2419
POWER APPROACH CONFIGURATION
C.G. - 32.43 % MAC
TOTAL PRESSURE SENSING FEEL SYSTEM
PICKUP LOCATED ON LEFT SIDE OF
VERTICAL FIN
STICK FORCE ~ LBS
BUFFET
MIN. GRADIENT
NORMAL ACCELERATION ~ g
1.0
1.4
1.8
2.2
2.6
3.0
3.4
3.8
STABILIZER DEFLECTION ~ DEG
NORMAL ACCELERATION ~ g

FIG. NO. 73

MANEUVERING FLIGHT CHARACTERISTICS

F-101A USAF No 53-2419

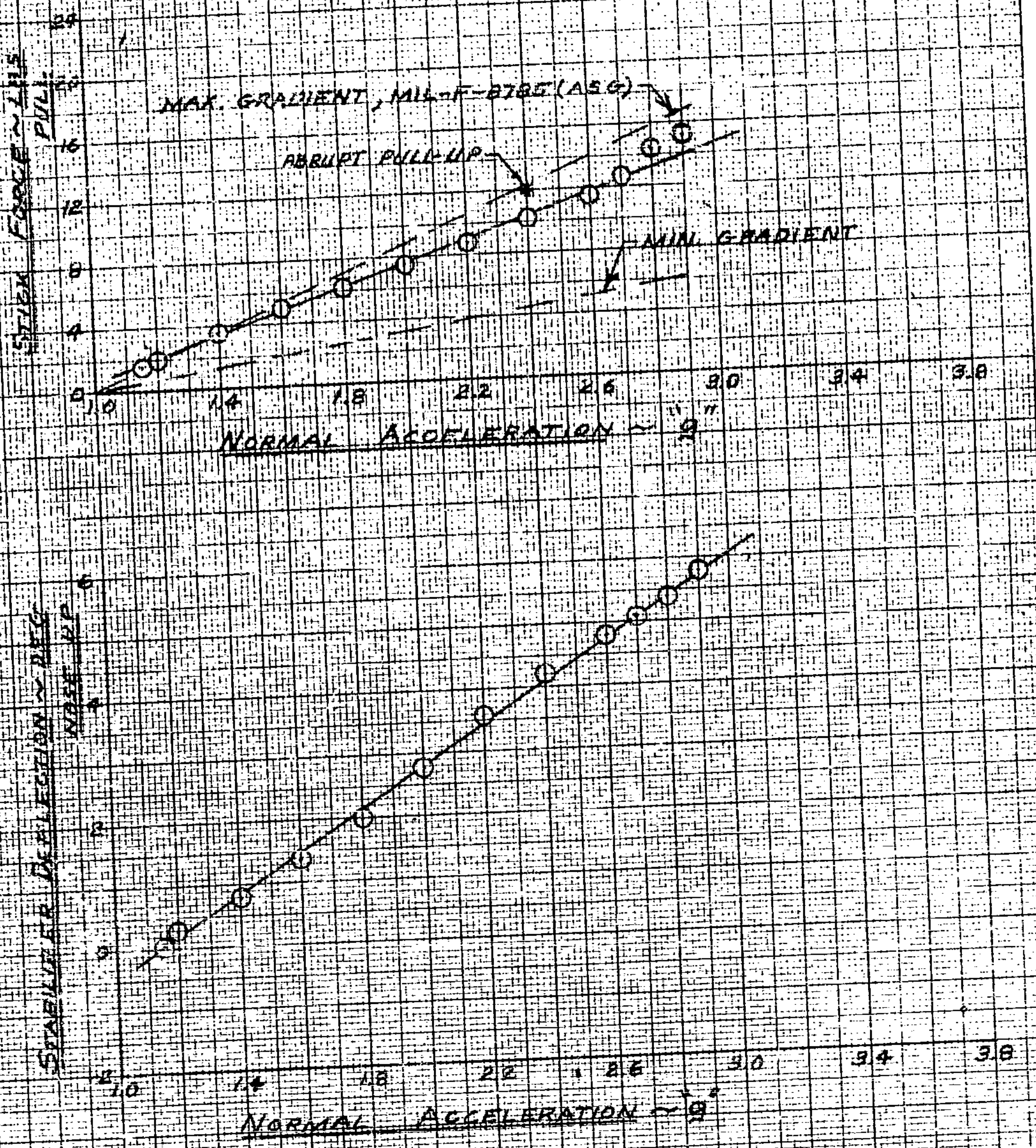
COMBAT CONFIGURATION

ALTITUDE – 35,000 FT

CAS – 474.5 KNOTS, MACH NO 1.309

GROSS WT – 32,050 LBS C.G. – 33.9 % MAC

TOTAL PRESSURE SENSING FEEL SYSTEM PICKUP LOCATED ON LEFT SIDE OF VERTICAL FIN.



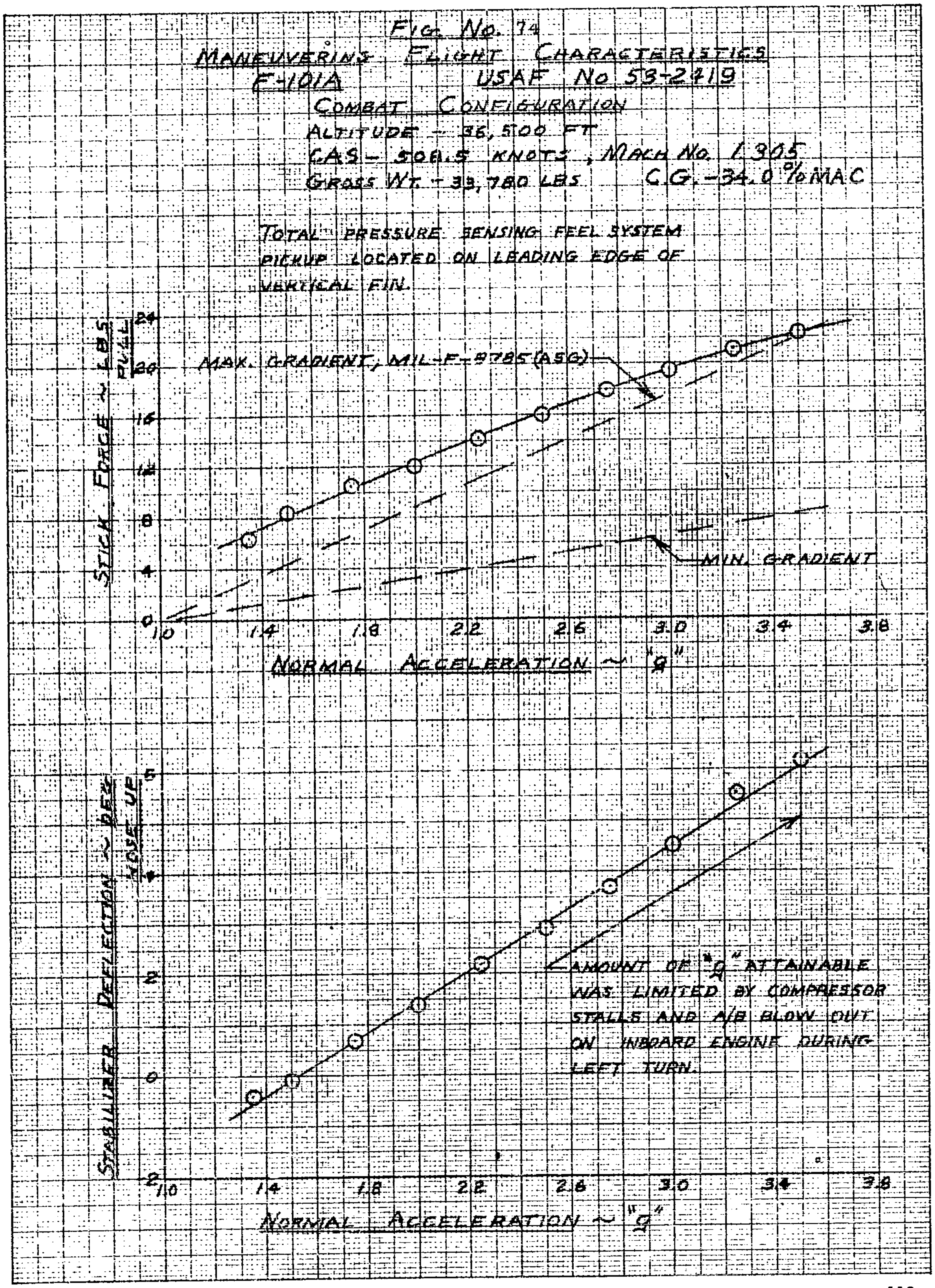
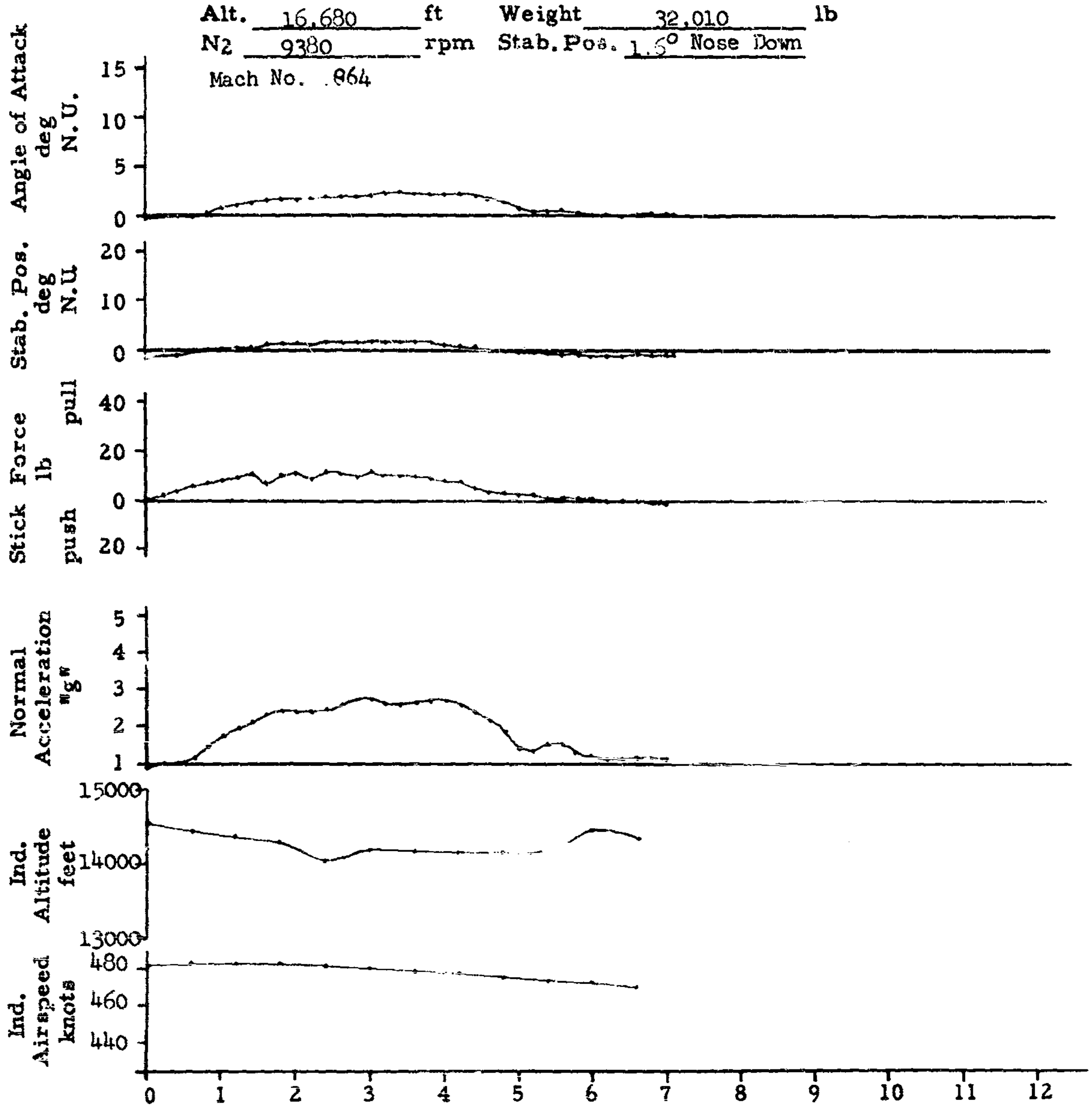


FIG. NO. 74
MANEUVERING FLIGHT CHARACTERISTICS
F-101A USAF No 53-2419
COMBAT CONFIGURATION
ALTITUDE - 36,500 FT
CAS - 508.5 KNOTS, MACH NO. 1.305
GROSS WT. - 33,780 LBS C.G. - 34.0 % MAC
TOTAL PRESSURE SENSING FEEL SYSTEM PICKUP LOCATED ON LEADING EDGE OF VERTICAL FIN.
STICK FORCE ~ LBS PULL
24
20
16
12
8
4
0
MAX. GRADIENT, MIL-F-8785 (ASG)
MIN. GRADIENT
1.0
1.4
1.8
2.2
2.6
3.0
3.4
3.8
NORMAL ACCELERATION ~ "g"
STABILIZER DEFLECTION ~ DEG NOSE UP
6
4
2
0
2
AMOUNT OF "g" ATTAINABLE WAS LIMITED BY COMPRESSOR STALLS AND A/B BLOW OUT ON INBOARD ENGINE DURING LEFT TURN.
1.0
1.4
1.8
2.2
2.6
3.0
3.4
3.8
NORMAL ACCELERATION ~ "g"

Figure No. 75
TIME HISTORY OF AN ABRUPT PULL-UP*
F-101A, USAF No. 53-2419
Configuration CRUISE

TRIM CONDITIONS

CAS 484 kts   CG pos. 33.3 %MAC
Alt. 16,680 ft   Weight 32,010 lb
N_2 9380 rpm   Stab. Pos. 1.5° Nose Down
Mach No. .864



Time - seconds
APPENDIX I

Figure No. 76

TIME HISTORY OF AN ABRUPT PULL-UP

F-101A, USAF No. 53-2419

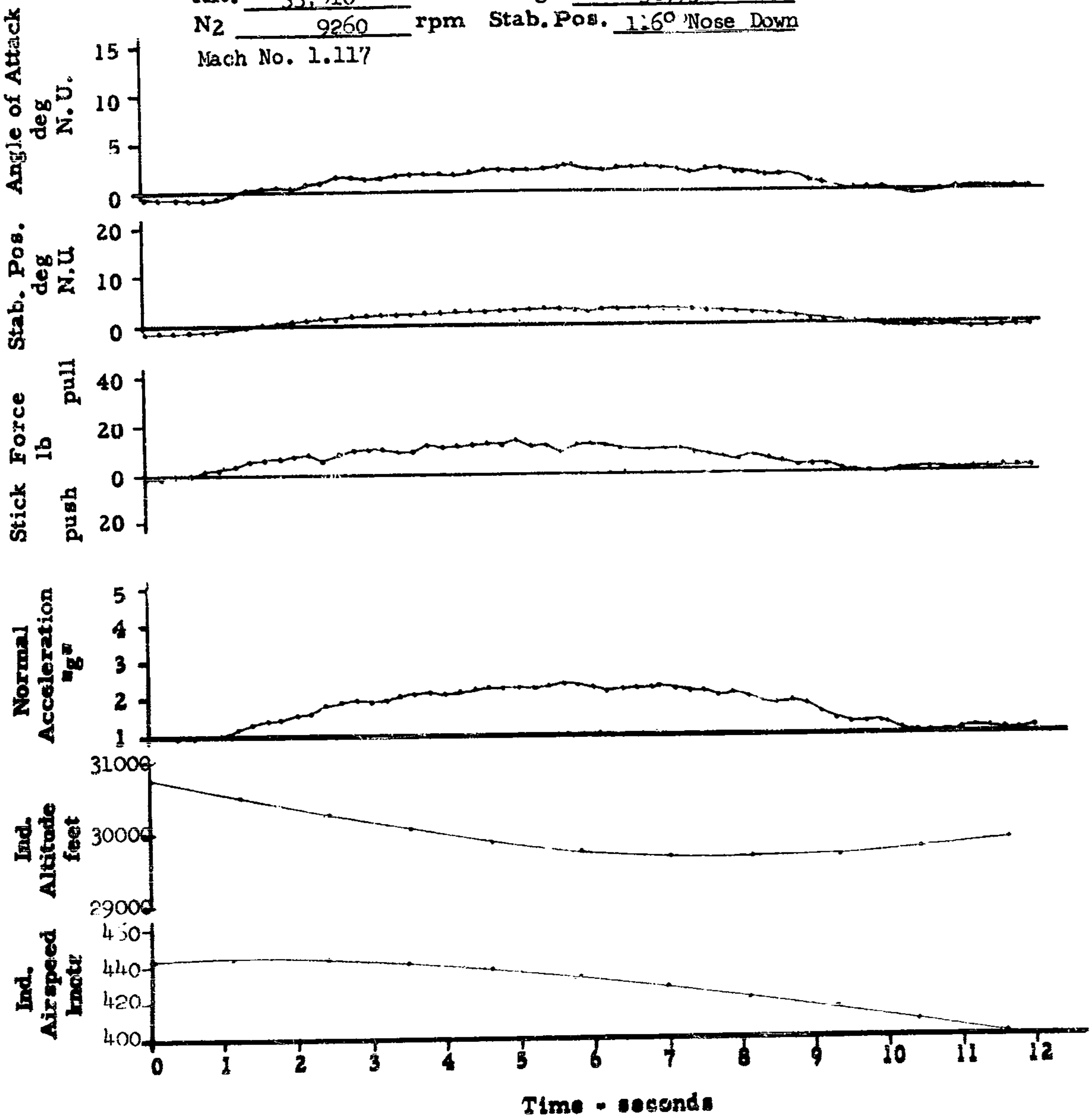
Configuration Combat

TRIM CONDITIONS

CAS 410 kts CG pos. 32.5 %MAC

Alt. 33,610 ft Weight 31,530 lb

N2 9260 rpm Stab. Pos. 1.6° Nose Down



APPENDIX I

Figure No. 77

TIME HISTORY OF SPEED BRAKE OPENING

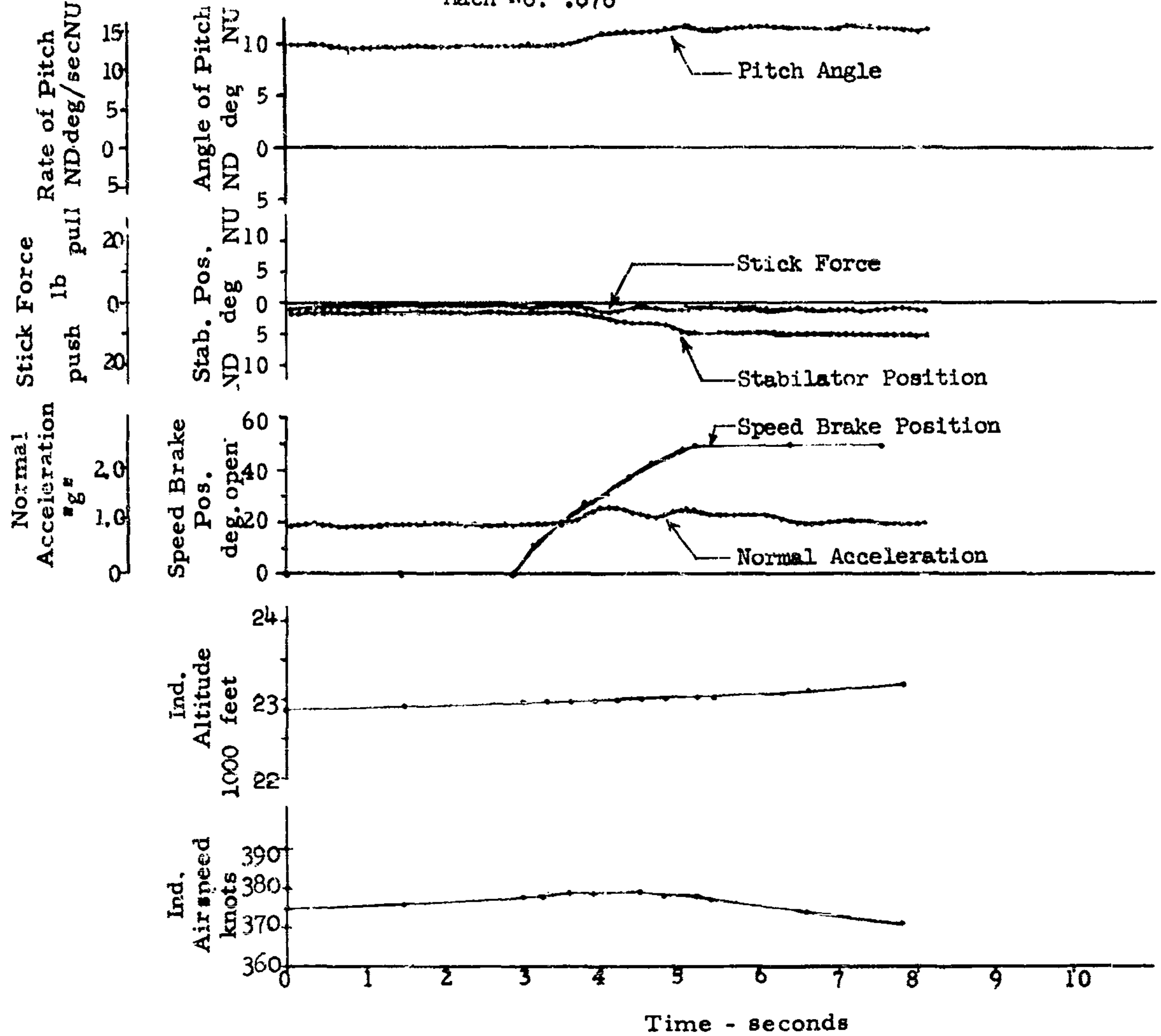
F-101A, USAF No. 53-2419

Configuration CRUISE

TRIM CONDITIONS

CAS 383 knots
Altitude 23,570 feet
Avg. N_2 9070 RPM
Mach No. .576

CG 32.1 %MAC
Weight 36,790 lb
Stab. Pos. 1.0 deg. A.N.D.



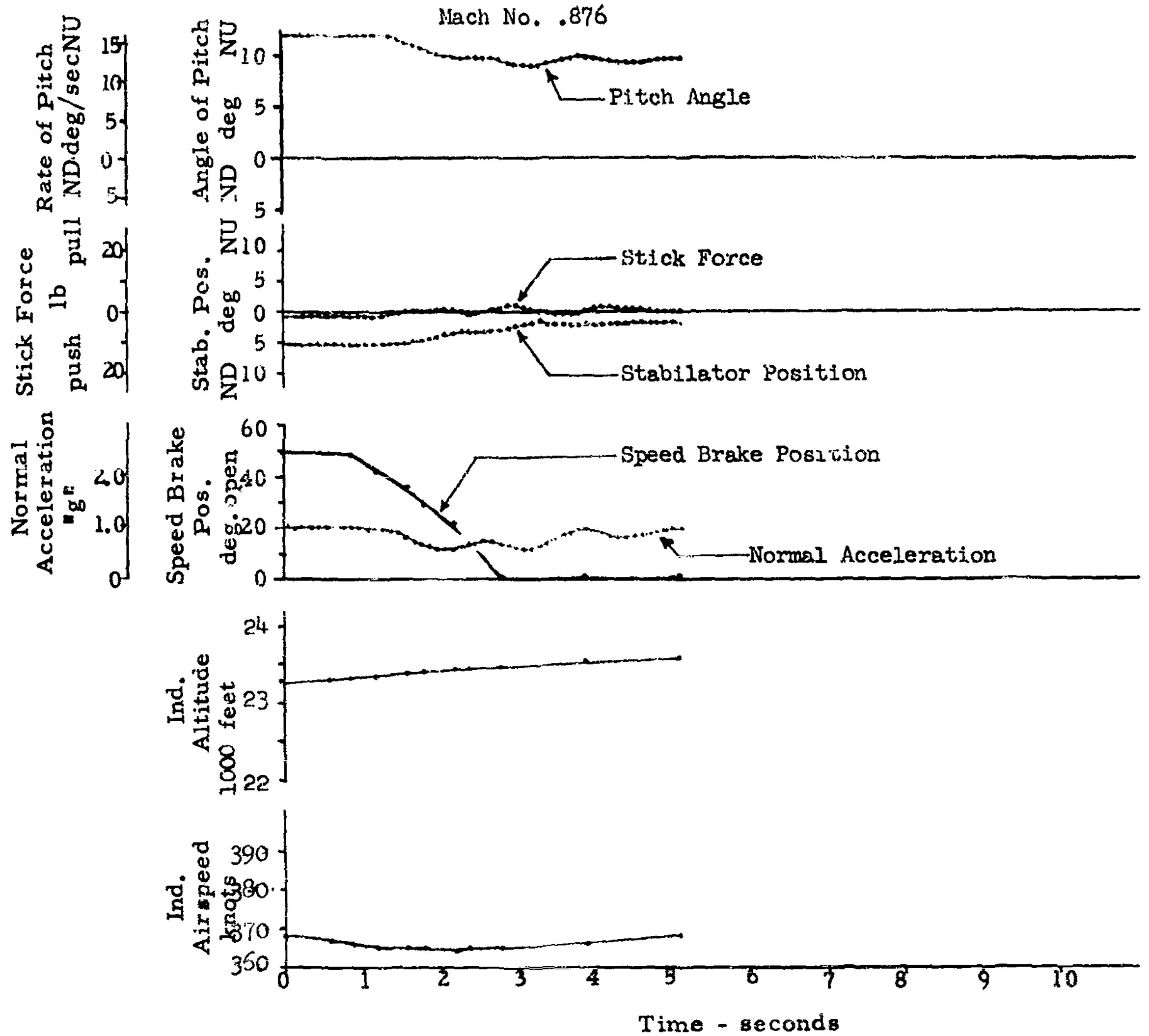
APPENDIX I

Figure No. 78
TIME HISTORY OF SPEED BRAKE CLOSING
F-101A, USAF No. 53-2419
Configuration CRUISE

TRIM CONDITIONS

CAS 383 knots		CG 32.1	%MAC
Altitude 23,570 feet		Weight 36,790	lb
Avg. N_2 9070 RPM		Stab. Pos. 1.0	deg. A.N.D.

Mach No. .876



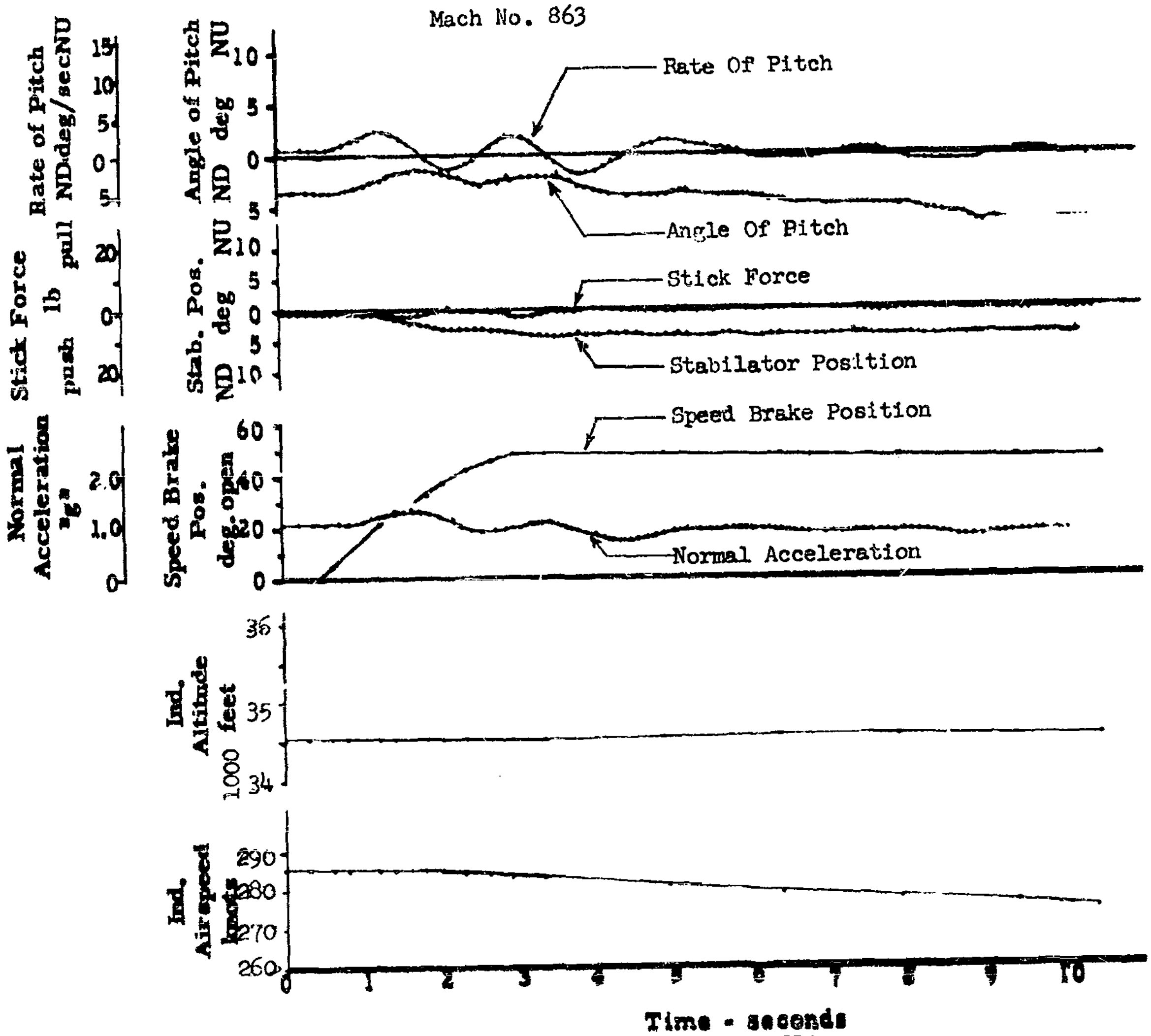
APPENDIX I

Figure No. 79
TIME HISTORY OF SPEED BRAKE OPENING
F-101A, USAF No. 53-2419
Configuration CRUISE

TRIM CONDITIONS

CAS 293 knots		CG 34.0 %MAC	
Altitude 35,390 feet		Weight 33,440 lb	
Avg. N_2 8340 RPM		Stab. Pos. 0 deg.	

Mach No. 863

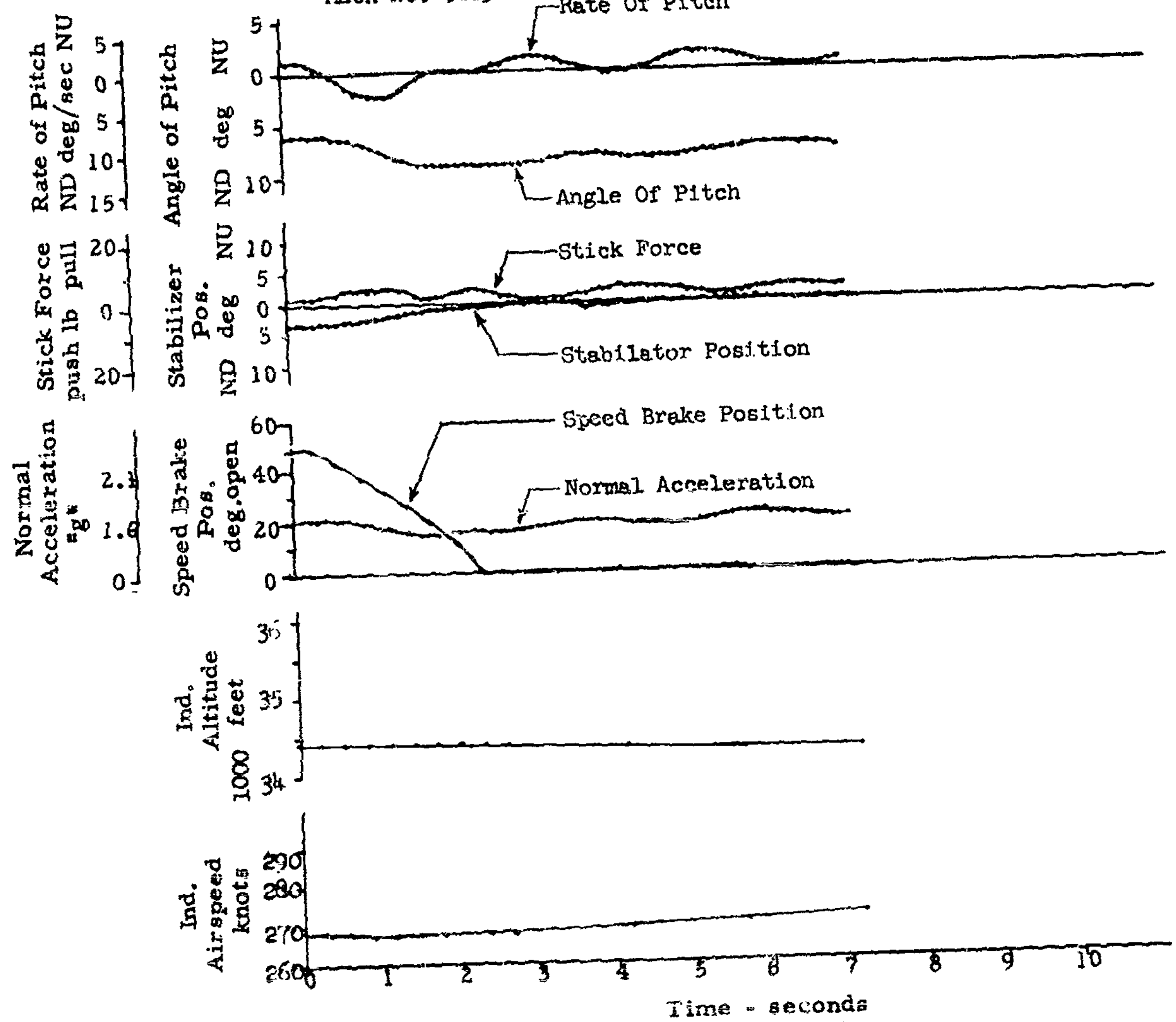


APPENDIX I

Figure No. 80

TIME HISTORY OF SPEED BRAKE CLOSING
F-101A, USAF No. 53-2419
Configuration CRUISE
TRIM CONDITIONS

CAS 293 knots
Altitude 35,390 feet
Avg. N_2 8340 RPM
CG 34.0 %MAC
Weight 33,440 lb
Stab. Pos. 0 deg.
Mach No. .863



APPENDIX I

Figure No. 81

TIME HISTORY OF SPEED BRAKE OPENING*

F-101A, USAF No. 53-2419

Configuration Combat

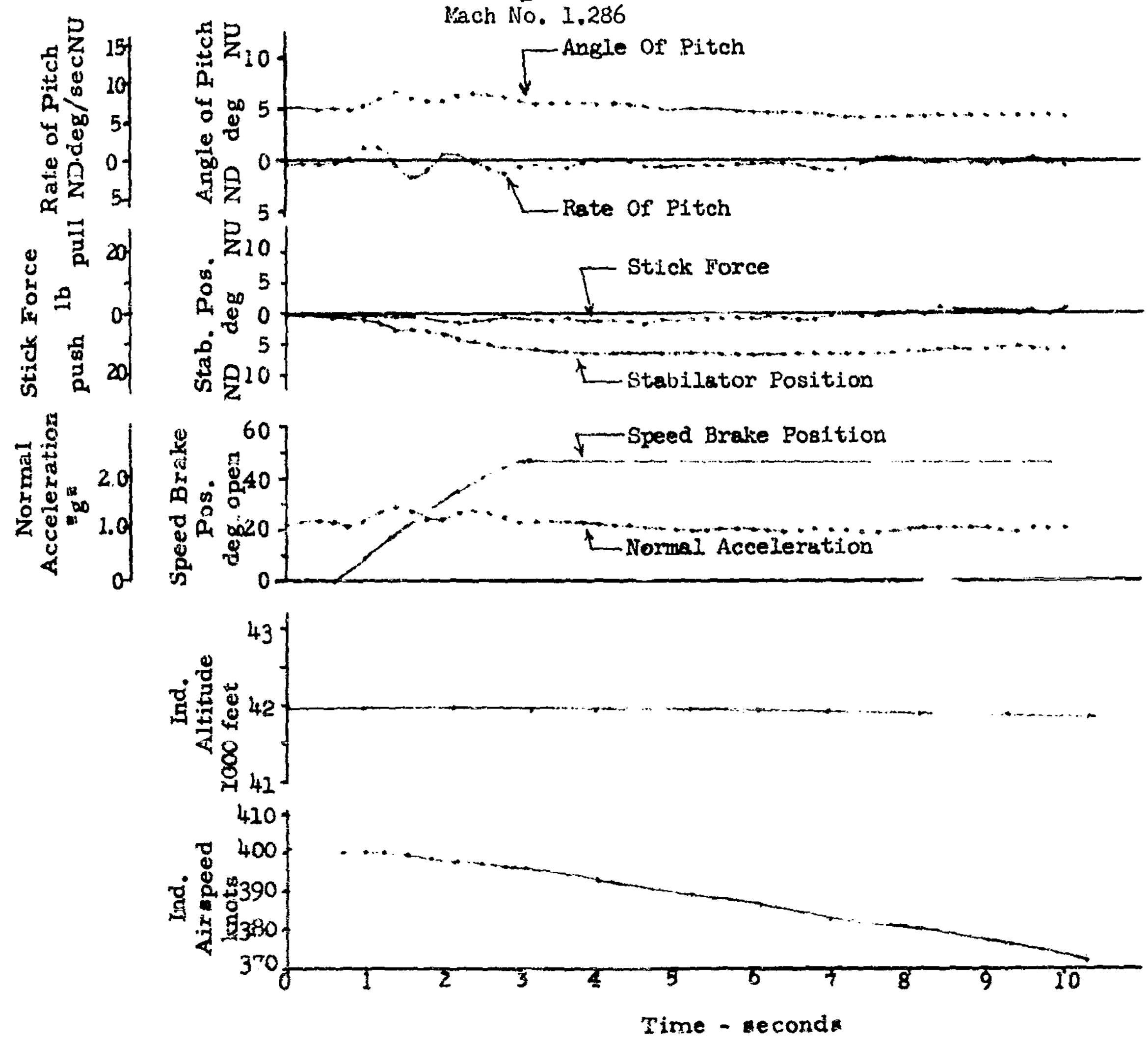
TRIM CONDITIONS

CAS 401 knots CG 33.6 %MAC

Altitude 41,930 feet Weight 32,140 lb

Avg. N_2 9000 RPM Stab. Pos. .7 deg. A.N.U.

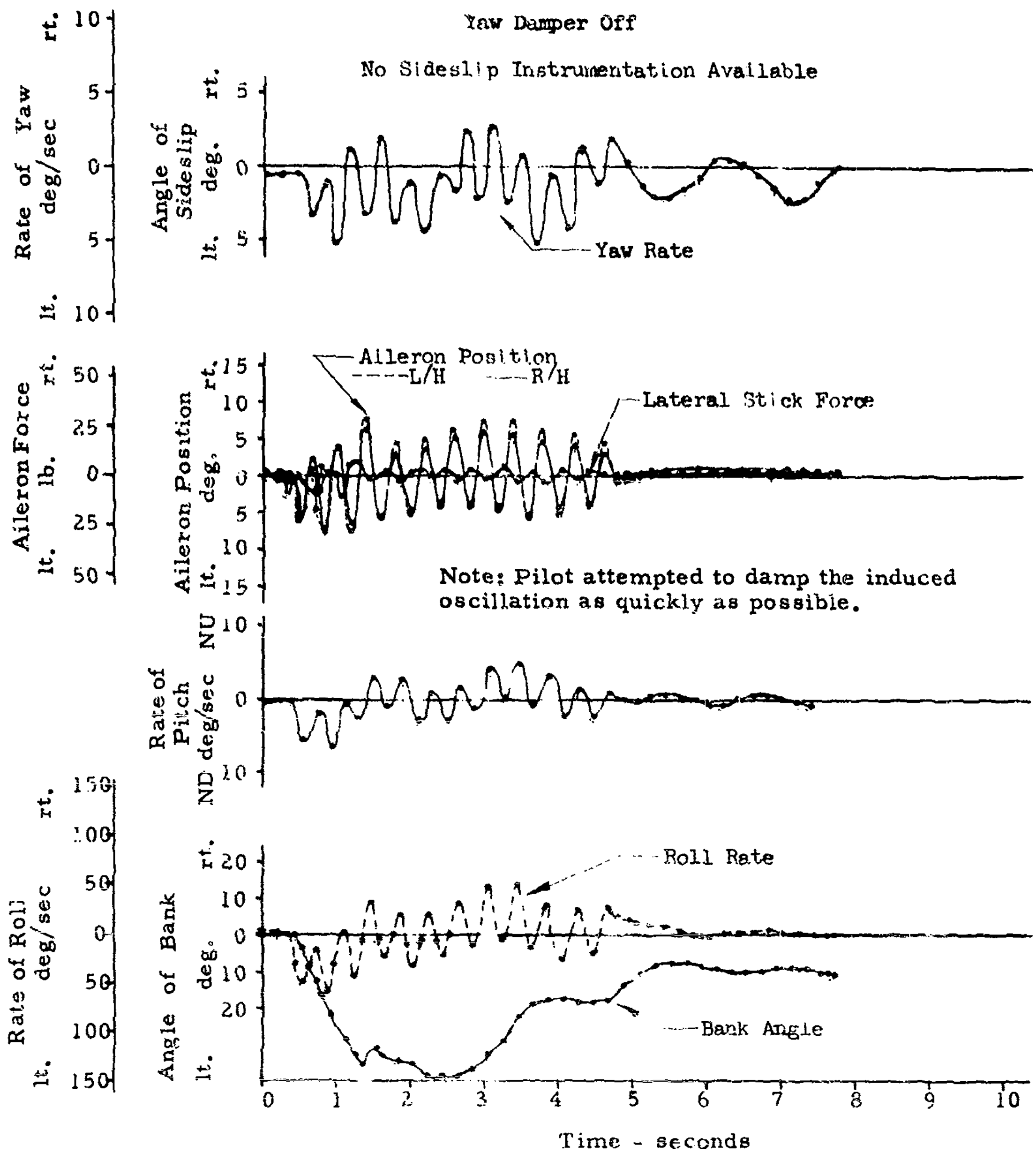
Mach No. 1.286



APPENDIX I

Figure No 82
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 433.5 knots		CG 34.5	%MAC
Altitude 11,440 feet		Weight 30,960	lb
Avg. N_2 8740 RPM		Rudder pos. 0	deg.
Mach No. .794		Aileron pos. 0	deg.



APPENDIX I

Figure No. 83
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	263.5	knots	CG	33.7	%MAC
Altitude	38,490	feet	Weight	32,130	lb
Avg. N_2	8270	RPM	Rudder pos.	0	deg.
Mach No.	.836		Aileron pos.	0	deg.

Yaw Damper Off

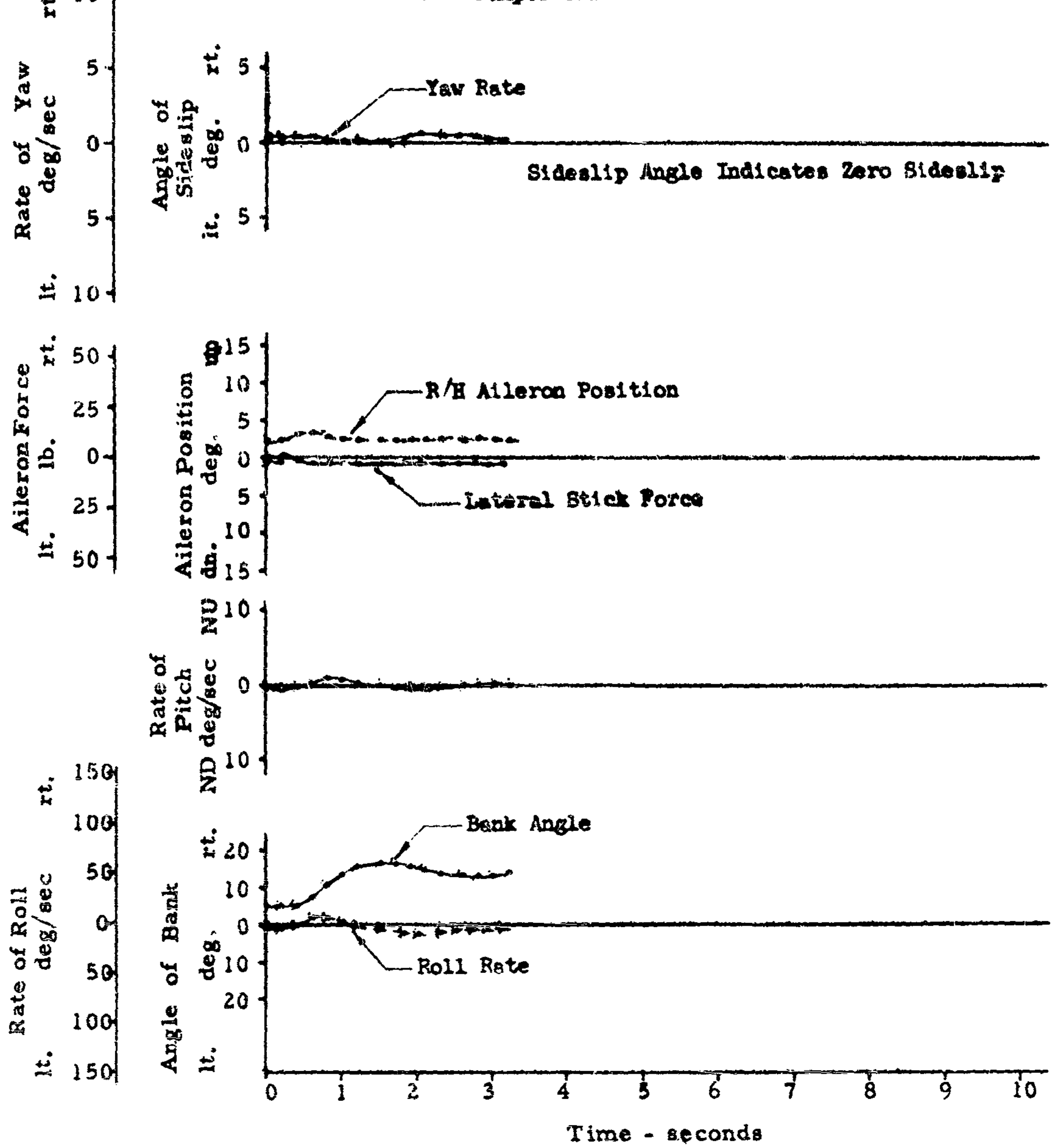


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Figure No. 84
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 263.5 knots		CG 33.7	%MAC
Altitude 38,490 feet		Weight 32,130	lb
Avg. N_2 8270 RPM		Rudder pos. 0	deg.
Mach No. .836		Aileron pos. 0	deg.

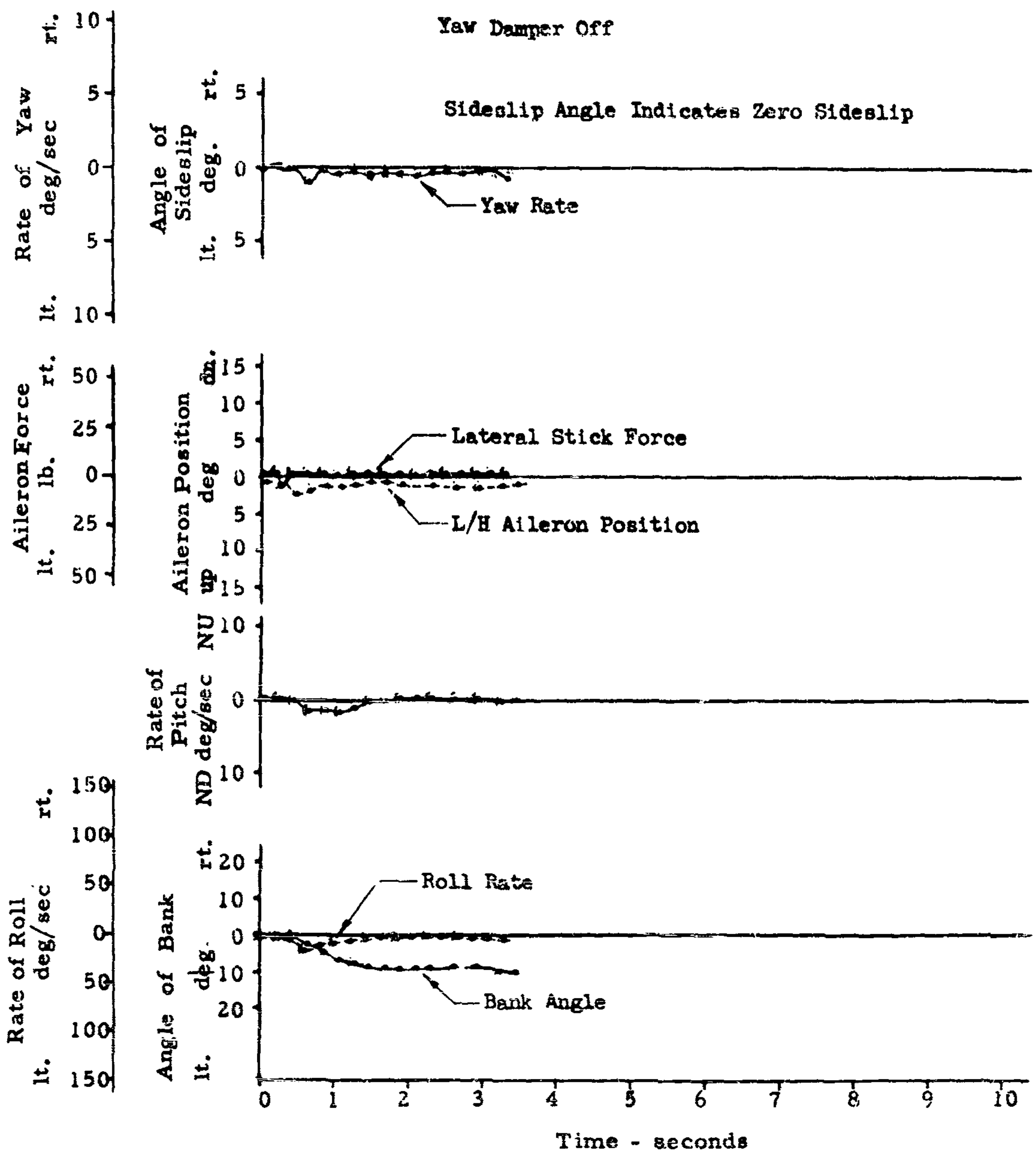

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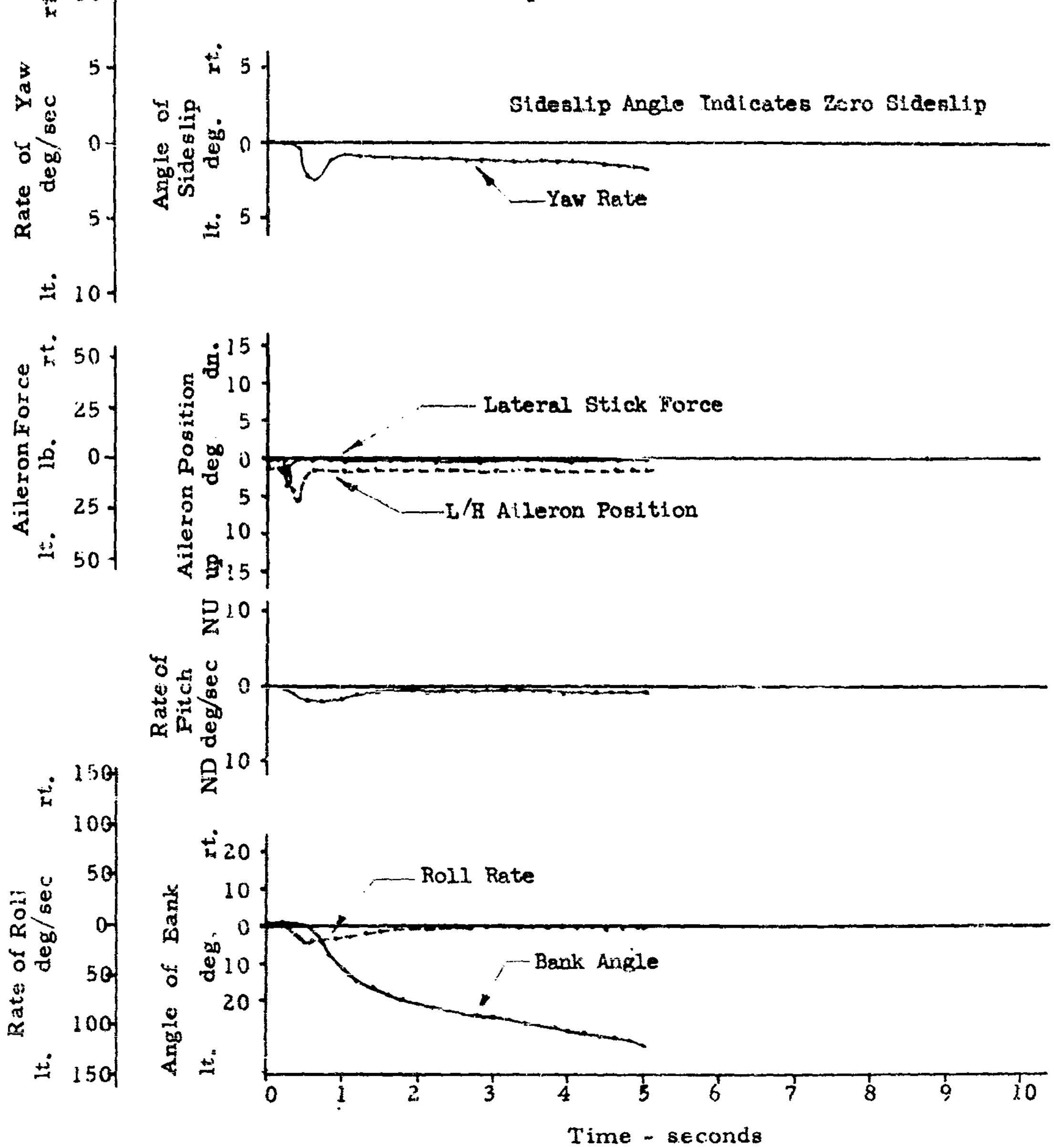
Figure No. 85
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 263.5 knots		CG 33.7	%MAC
Altitude 38,490 feet		Weight 32,130	lb
Avg. N_2 8270 RPM		Rudder pos. 0	deg.
Mach No. .836		Aileron pos. 0	deg.

Yaw Damper On

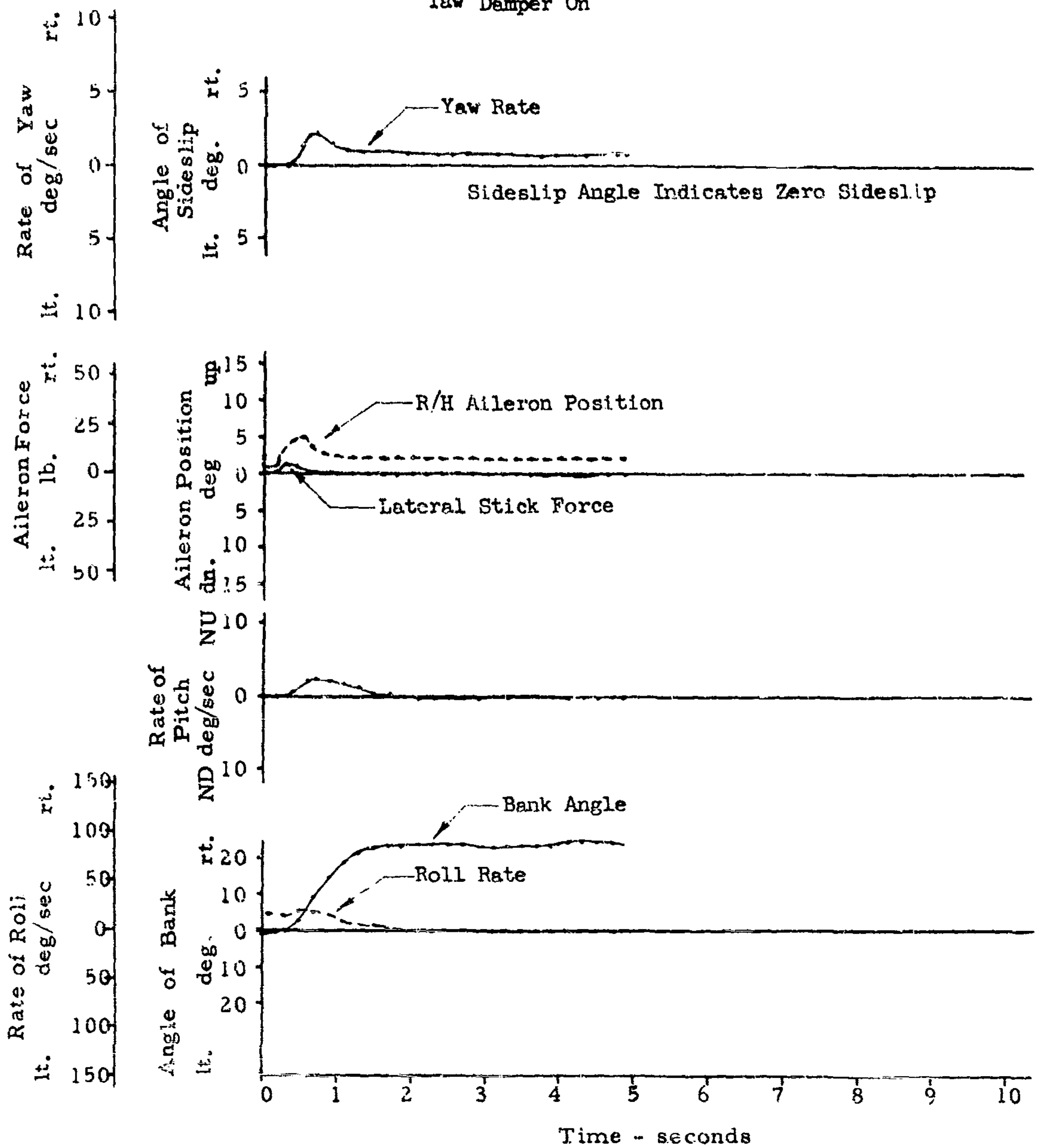


APPENDIX I

Figure No. 86
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 263.5 knots		CG 33.7	%MAC
Altitude 38,490 feet		Weight 32,130	lb
Avg. N_2 8270 RPM		Rudder pos. 0	deg.
Mach No. .836		Aileron pos. 0	deg.

Yaw Damper On

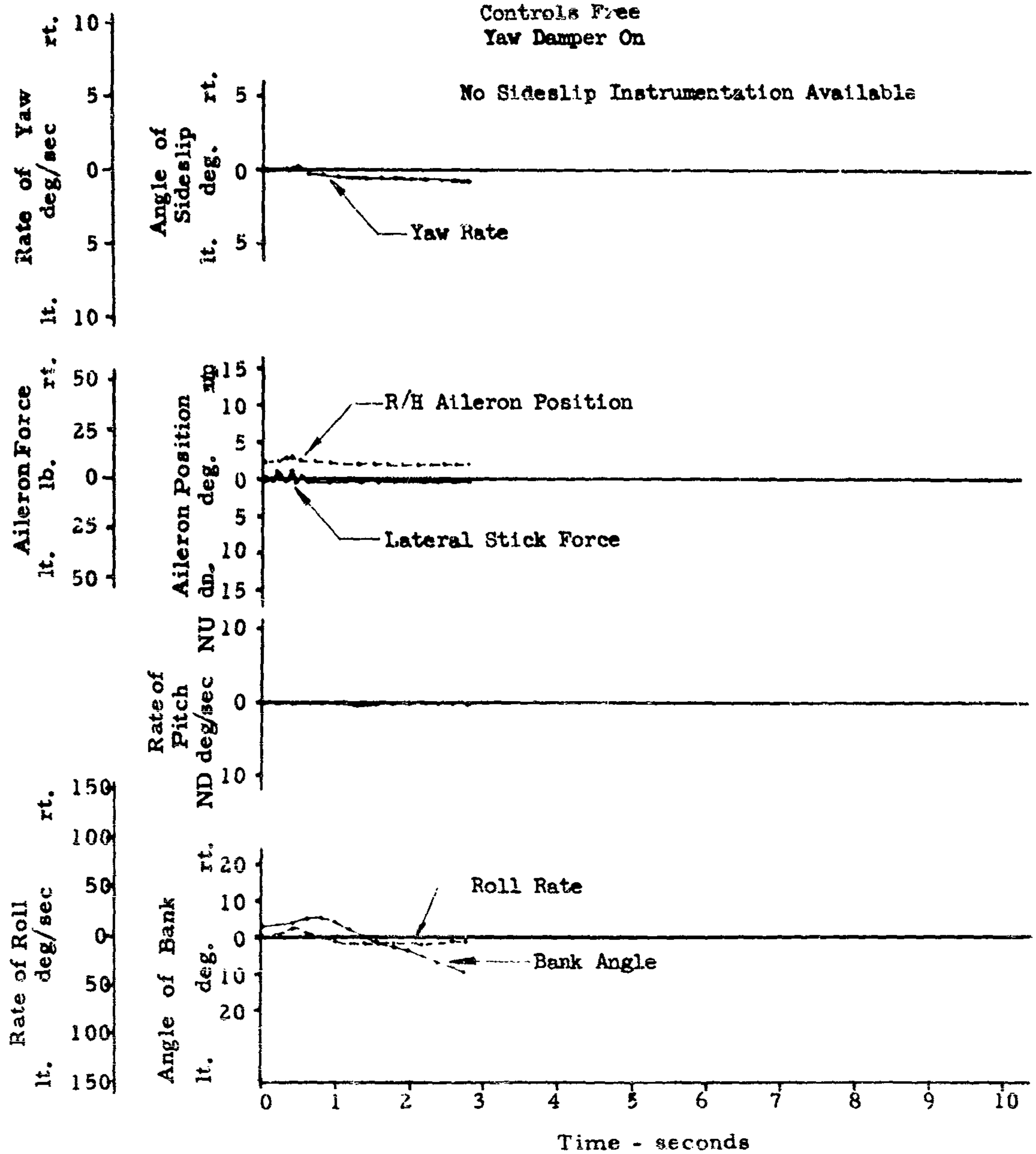


APPENDIX I

Figure No. 87
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 417 knots		CG 32.7	%MAC
Altitude 12,730 feet		Weight 38,570	lb
Avg. N_2 8900 RPM		Rudder pos. 1.0	deg. nose right
Mach No. .783		Aileron pos. 0	deg.

Controls Free
Yaw Damper On

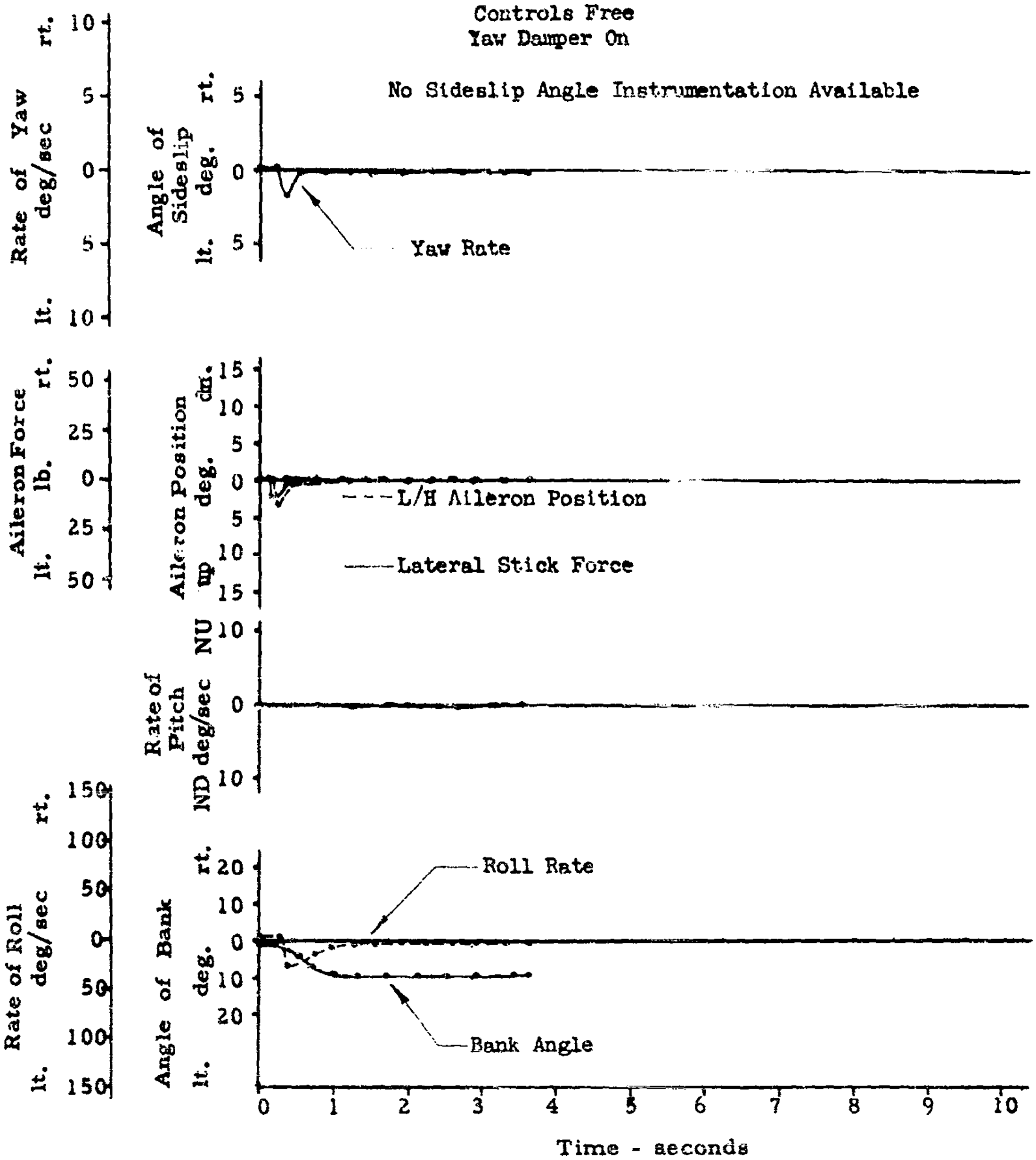


APPENDIX I

Figure No. 88
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 417 knots		CG 32.7	%MAC
Altitude 12,730 feet		Weight 38,570	lb
Avg. N_2 8900 RPM		Rudder pos. 1.0	deg. nose right
Mach No. .783		Aileron pos. 0	deg.

Controls Free
Yaw Damper On



APPENDIX I

Figure No. 89
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 417 knots	CG	32.7	%MAC	
Altitude 12,730 feet	Weight	38,570	lb	
Avg. N_2 8900 RPM	Rudder pos.	1.0	deg. nose right	
Mach No. .783	Aileron pos.	0	deg.	

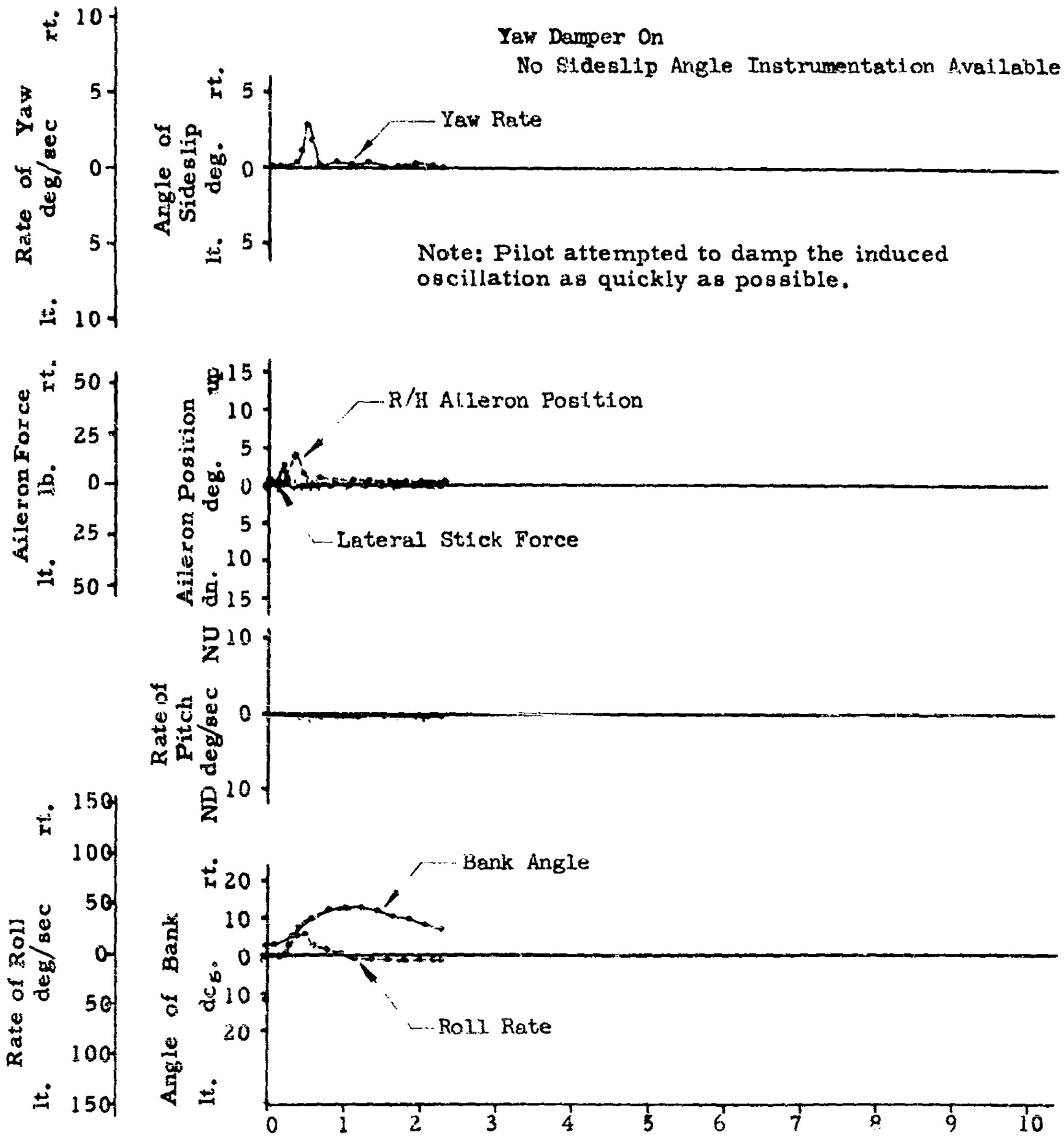


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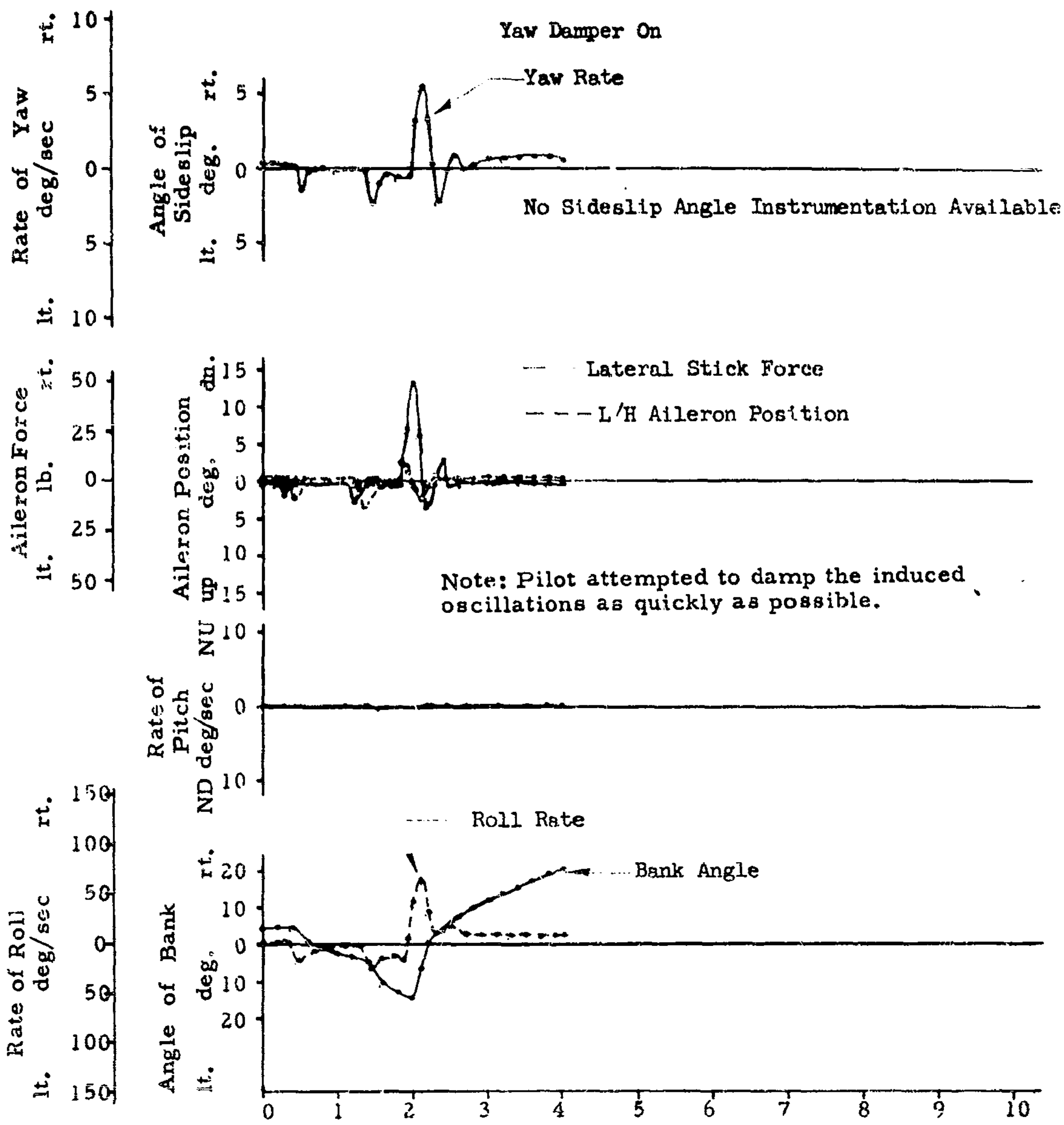
Figure No. 90
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	417 knots	CG	32.7 %MAC
Altitude	12,730 feet	Weight	38,570 lb
Avg. N_2	8900 RPM	Rudder pos.	1.0 deg. nose right
Mach No.	.783	Aileron pos.	0 deg.



APPENDIX I

Figure No. 91
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 417 knots — CG 32.7 %MAC
Altitude 12,730 feet — Weight 38,570 lb
Avg. N_2 8900 RPM — Rudder pos. 1.0 deg. nose right
Mach No. .783 — Aileron pos. 0 deg.

Controls Free
Yaw Damper Off

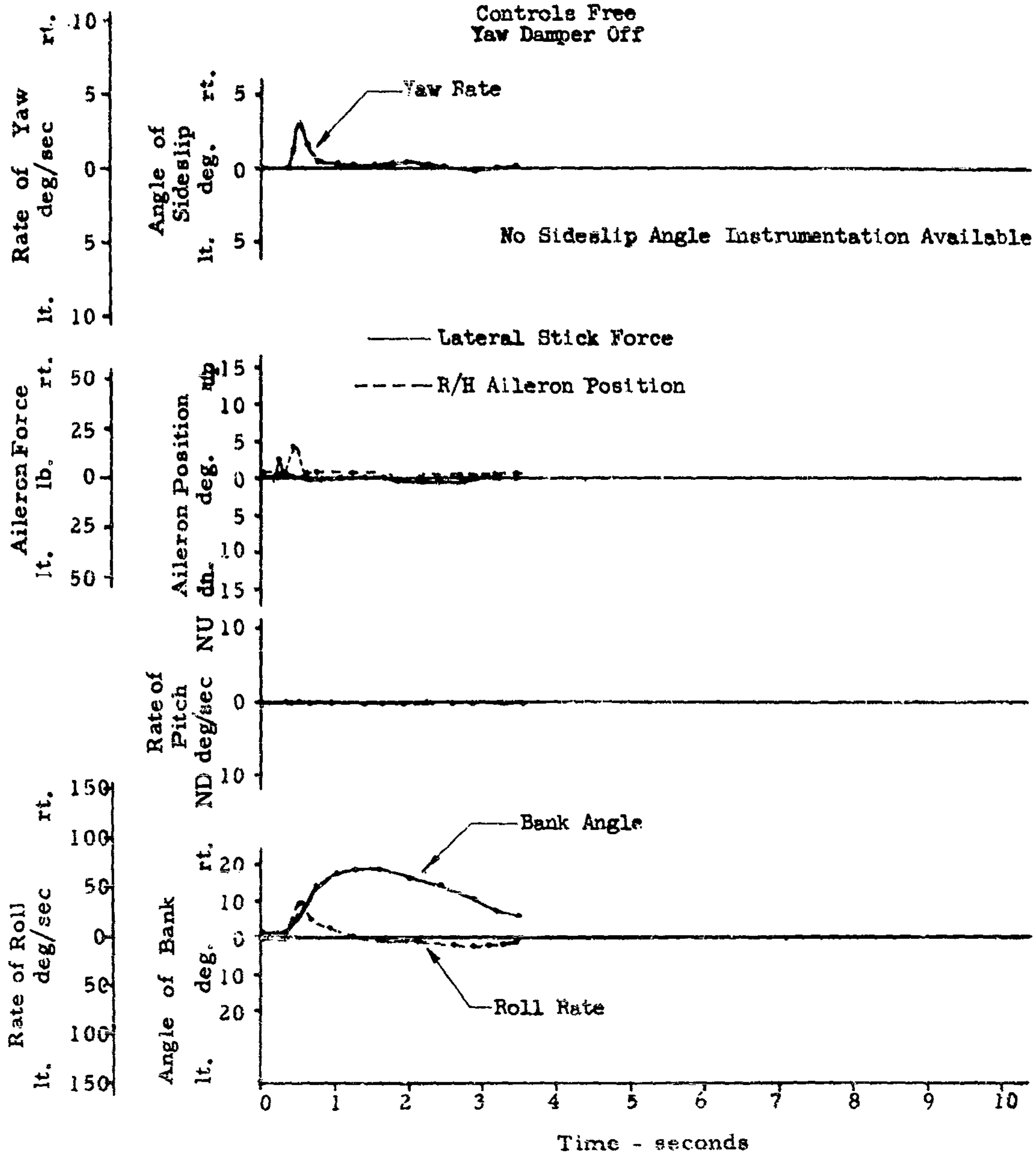


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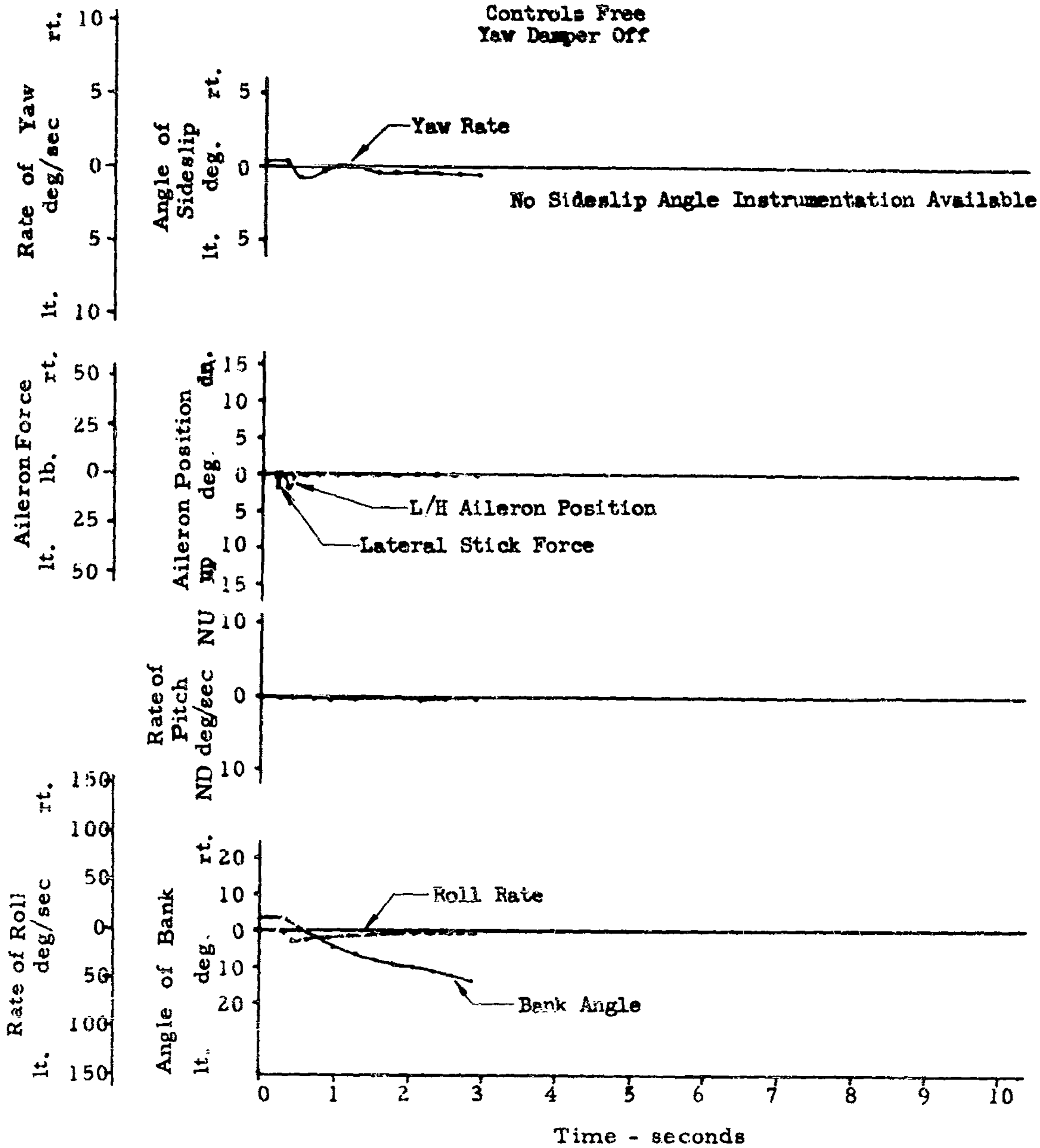
Figure No. 92
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 417 knots		CG 32.7 %MAC	
Altitude 12,730 feet		Weight 38,570 lb	
Avg. N_2 8900 RPM		Rudder pos. 1.0 deg. nose right	
Mach No. .783		Aileron pos. 0 deg.	

Controls Free
Yaw Damper Off



APPENDIX I

Figure No. 93
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots		CG 32.6	%MAC
Altitude 14,270 feet		Weight 35,860	lb
Avg. N_2 9380 RPM		Rudder pos. 1.4	deg. nose right
Mach No. .973		Aileron pos. .5	deg. left wing up

Controls Free
Yaw Damper On

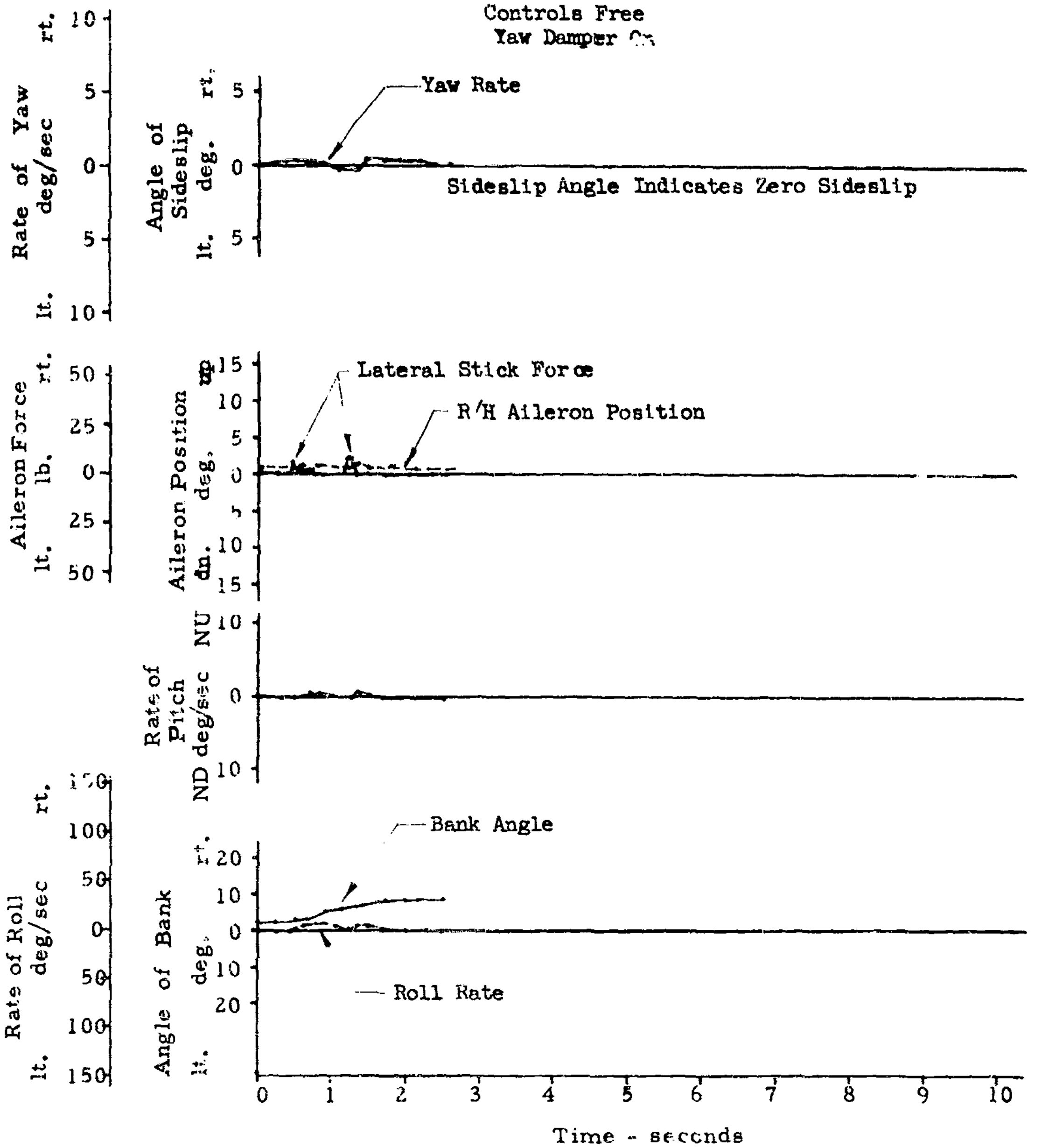


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Figure No. 94
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots		CG 32.6 %MAC	
Altitude 14,270 feet		Weight 35,860 lb	
Avg. N_2 9380 RPM		Rudder pos. 1.4 deg.	nose right
Mach No. .973		Aileron pos. .5 deg.	left wing up

Controls Free
Yaw Damper On

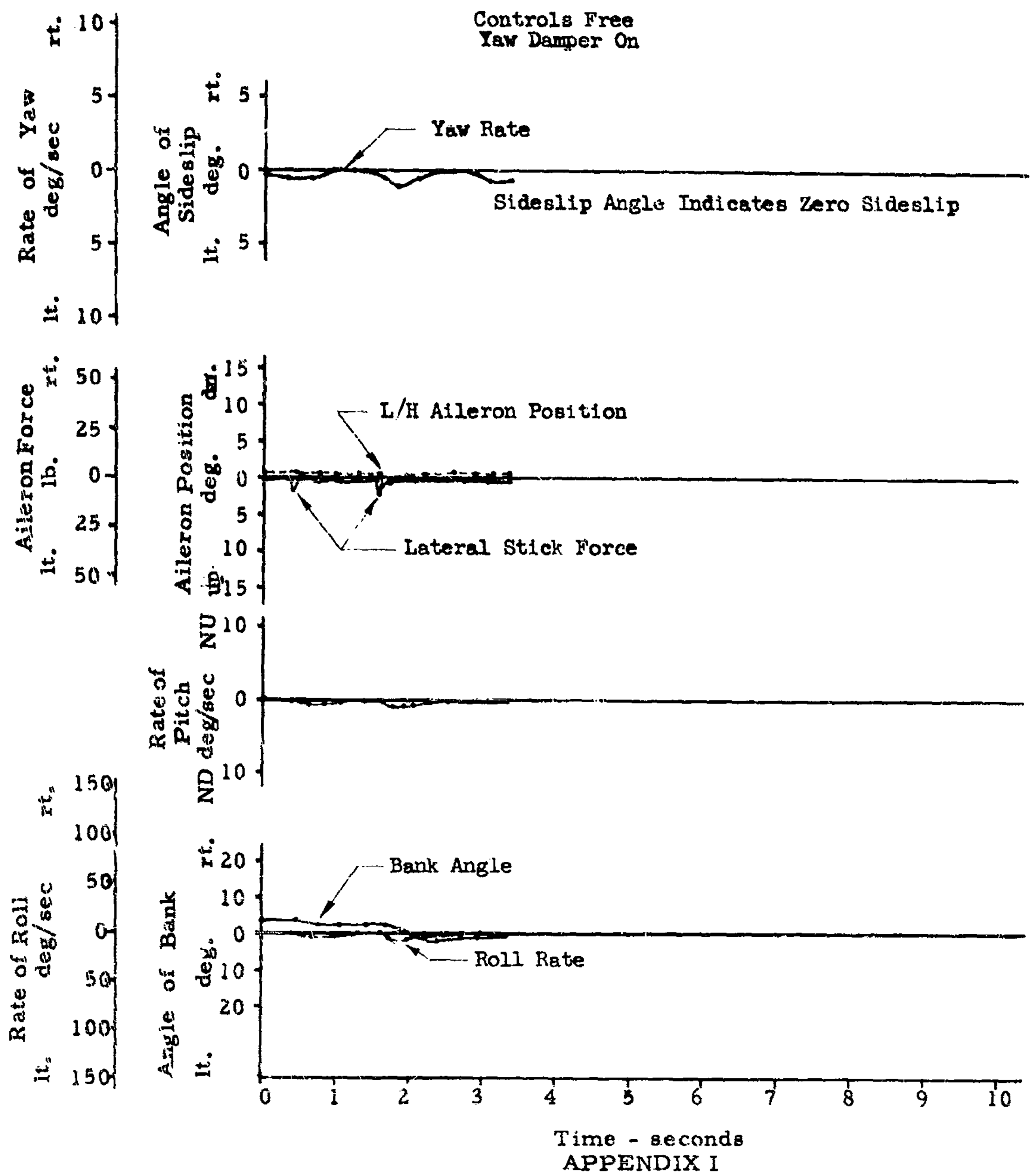

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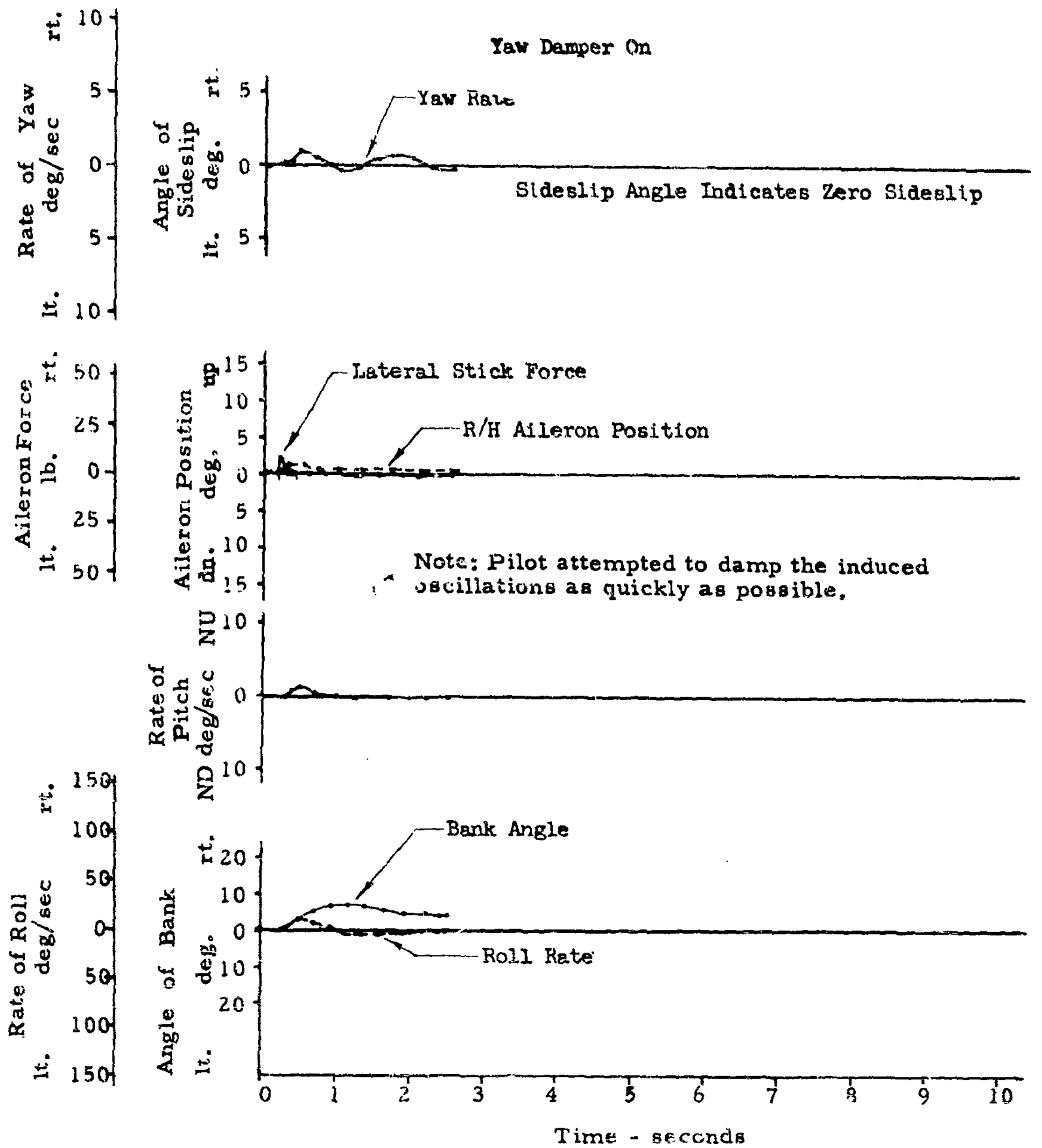
Figure No. 95
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots	CG 32.6 %MAC		
Altitude 14,270 feet	Weight 35,860 lb		
Avg. N_2 9380 RPM	Rudder pos. 1.4 deg.	nose right	
Mach No. .973	Aileron pos. .5 deg.	left wing up	



APPENDIX I

Figure No. 96

DYNAMIC LATERAL STABILITY

F-101A, USAF No. 53-2419

Combat Configuration

TRIM CONDITIONS

CAS	511 knots	CG	32.6	%MAC	
Altitude	14,270 feet	Weight	35,860	lb	
Avg. N_2	9380 RPM	Rudder pos.	1.4	deg.	nose right
Mach No. .973		Aileron pos.	.5	deg.	left wing up

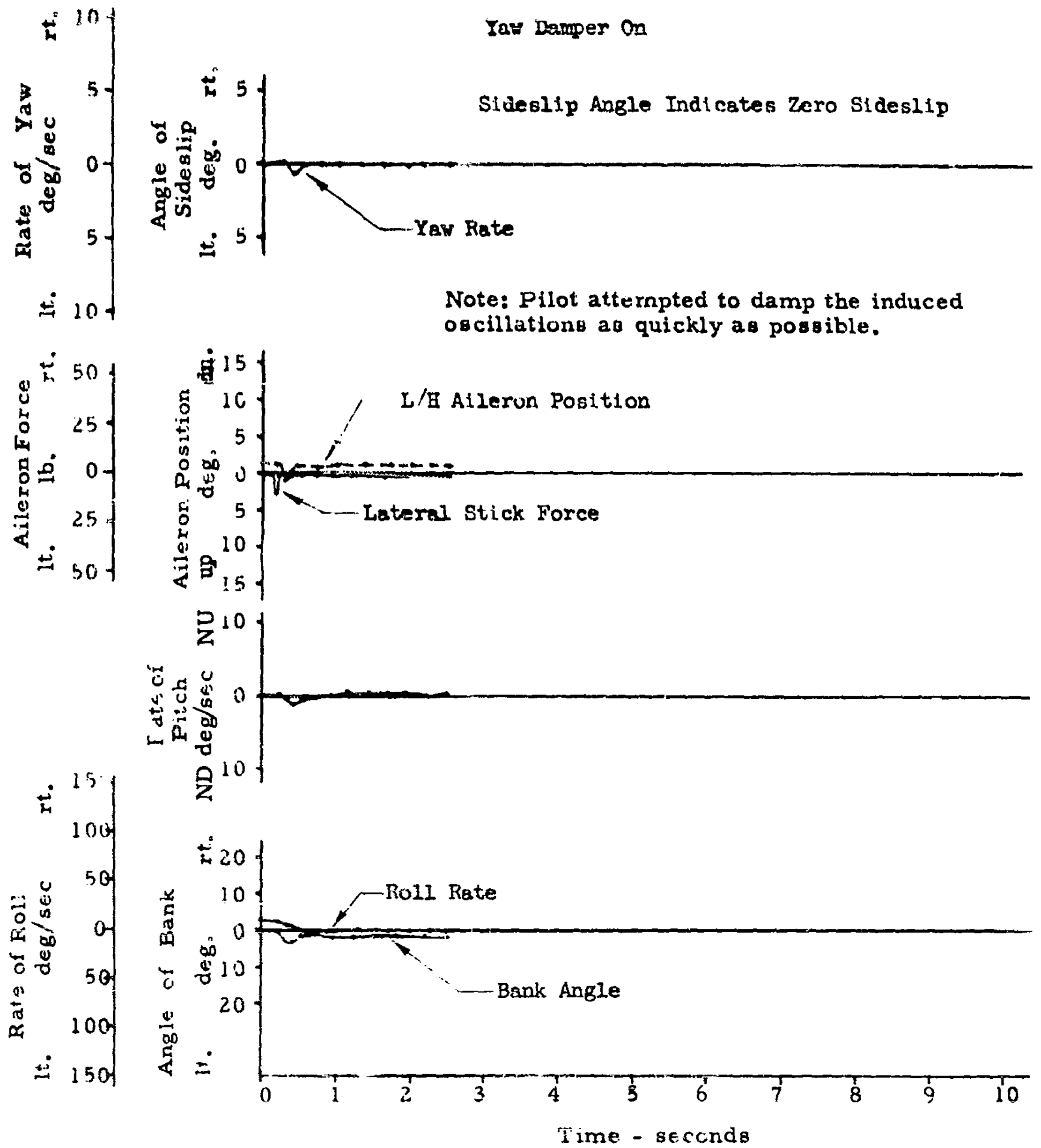


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Figure No. 97
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots		CG 32.6	%MAC
Altitude 14,270 feet		Weight 35,860	lb
Avg. N_2 9380 RPM		Rudder pos. 1.4	deg. nose right
Mach No. .973		Aileron pos. .5	deg. left wing up

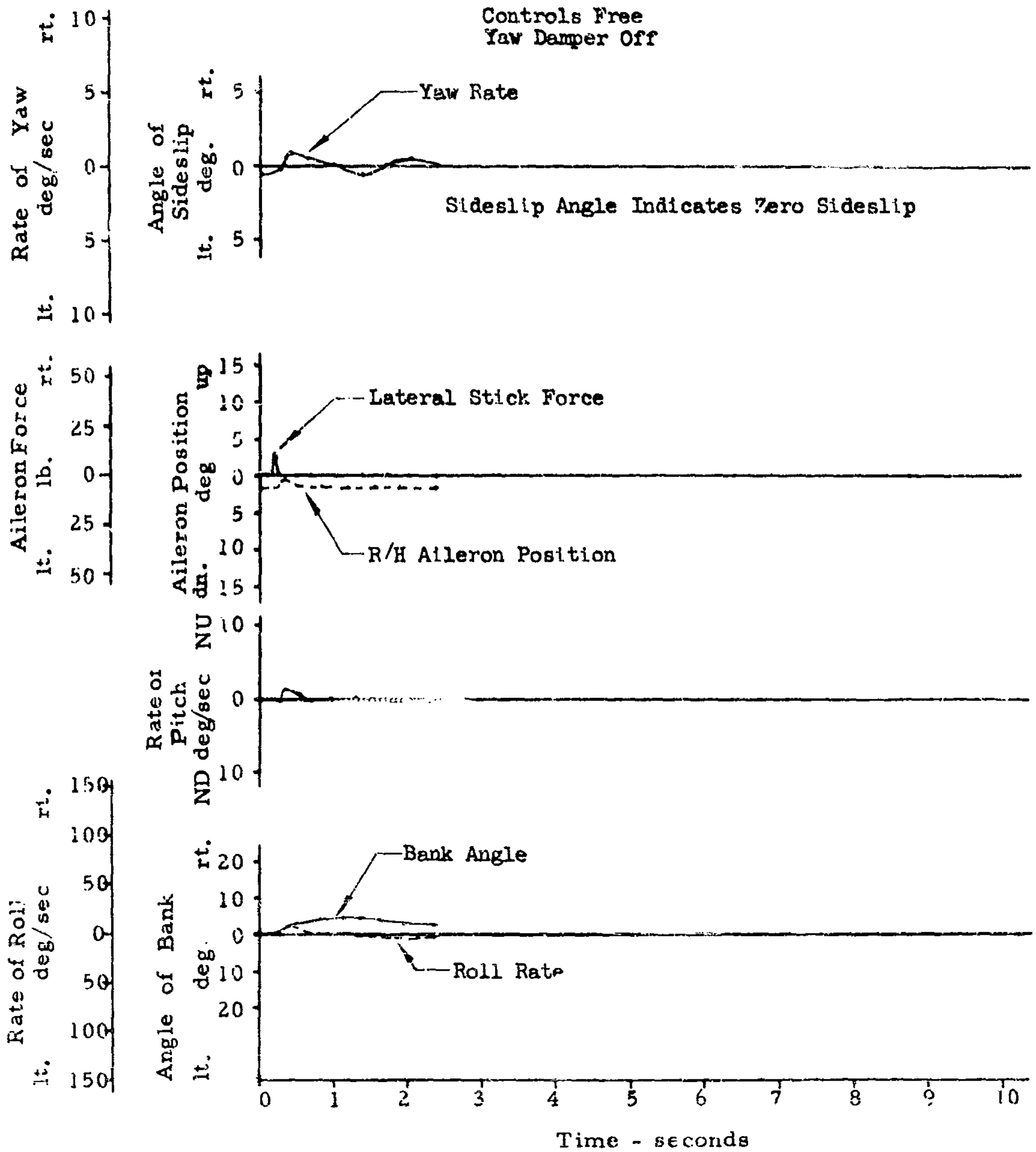

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Figure No. 98
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots	CG 32.6 %MAC		
Altitude 14,270 feet	Weight 35,860 lb		
Avg. N_2 9380 RPM	Rudder pos. 1.4 deg.	nose right	
Mach No. .973	Aileron pos. .5 deg.	left wing up	

Controls Free
Yaw Damper Off

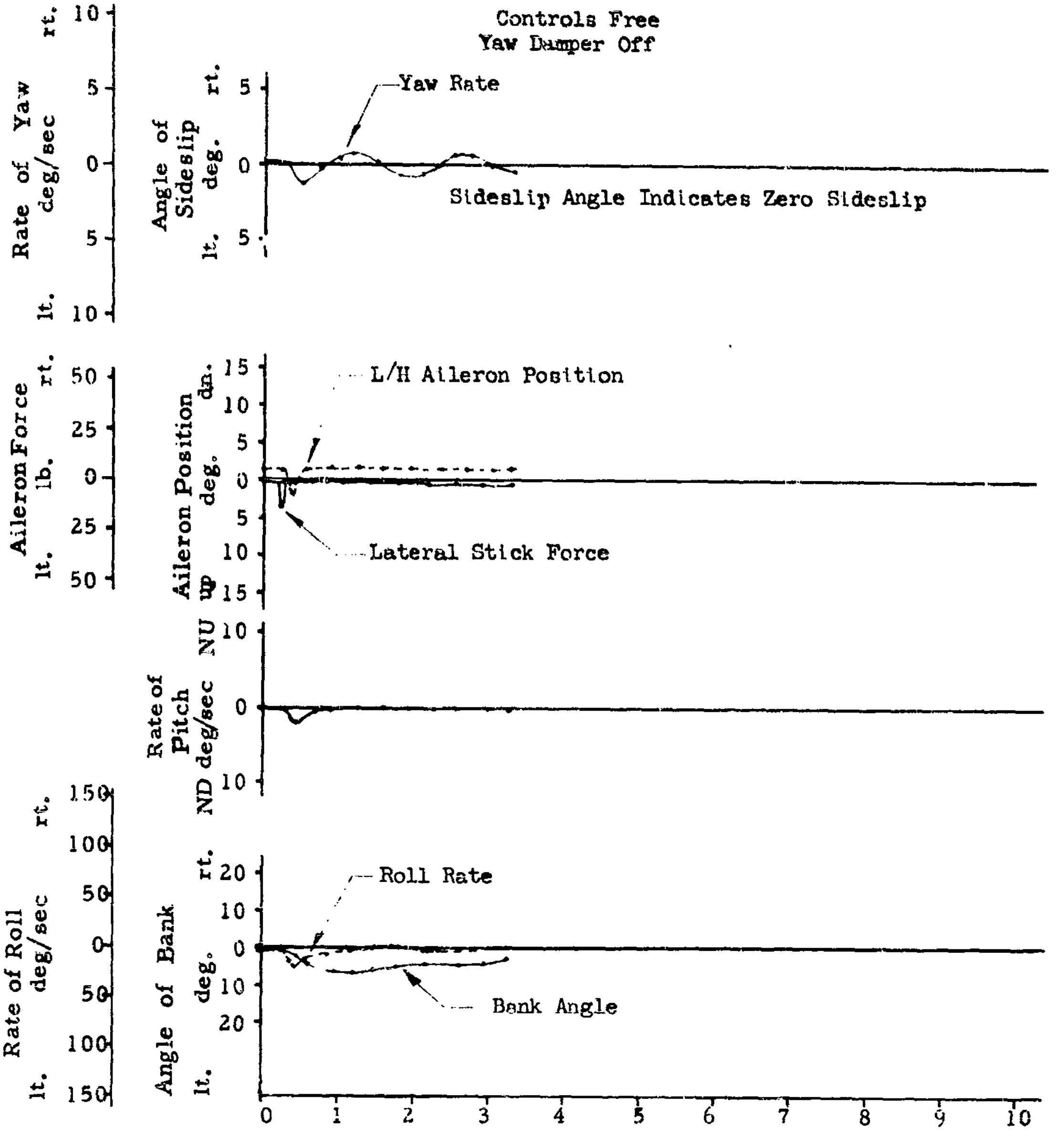

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Figure No. 99
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193 knots		CG 32.4	%MAC
Altitude 12,660 feet		Weight 31,750	lb
Avg. N_2 9150 RPM		Rudder pos. 0	deg.
		Aileron pos. 0	deg.

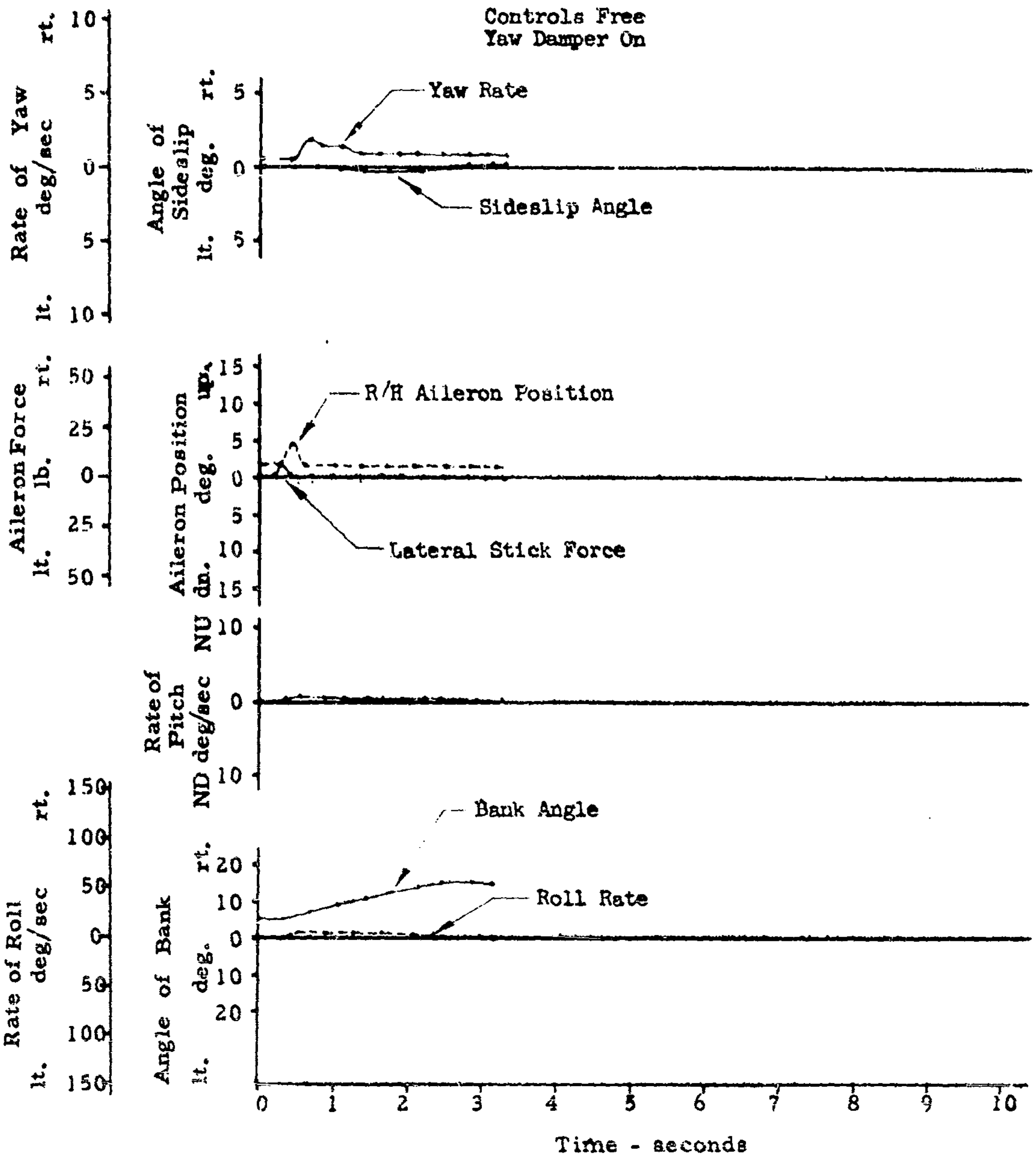

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Figure No. 100
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193 knots		CG 32.4	%MAC
Altitude 12,660 feet		Weight 31,750	lb
Avg. N_2 9150 RPM		Rudder pos. 0	deg.
		Aileron pos. 0	deg.

Controls Free
Yaw Damper On

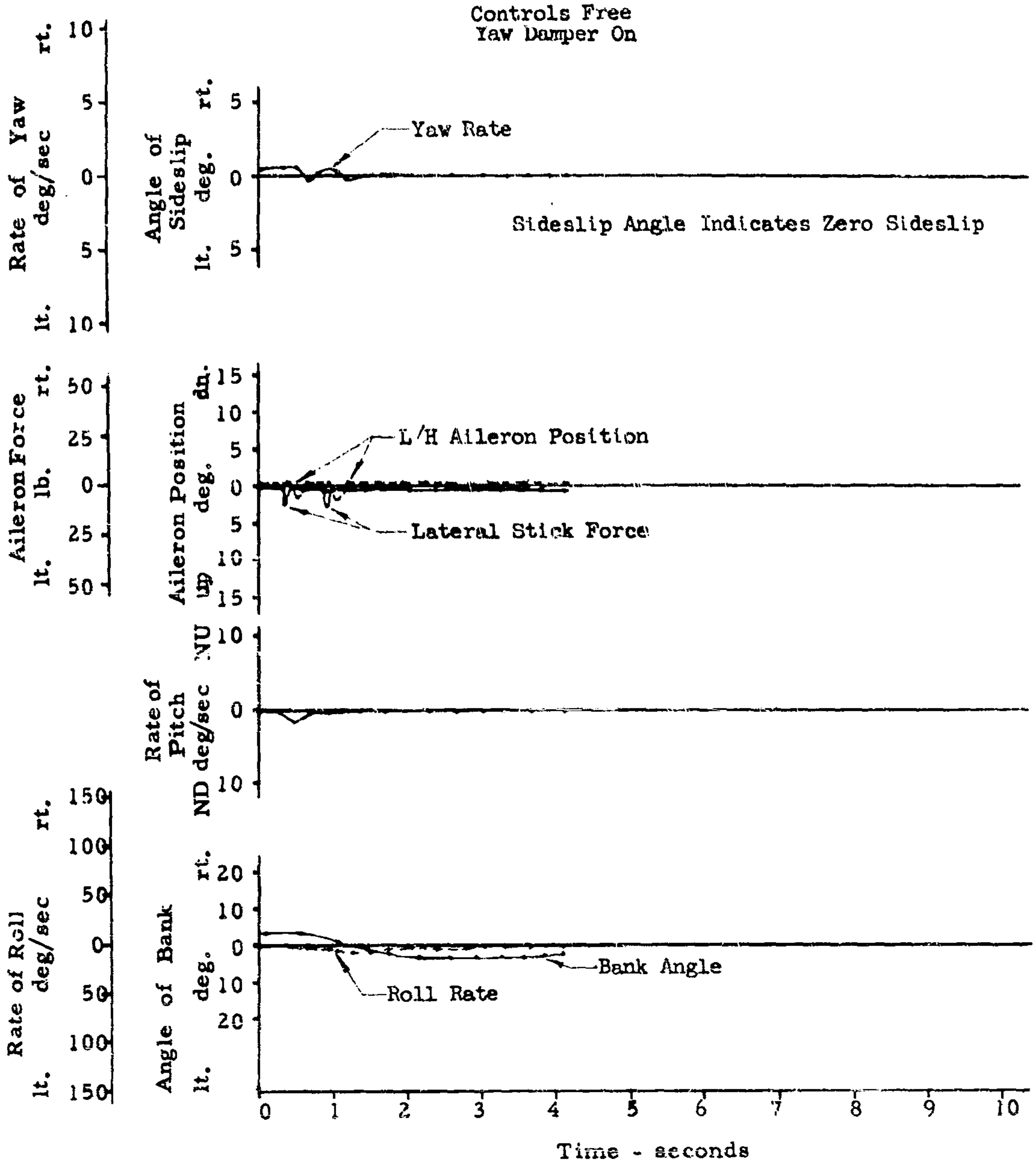

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Figure No. 101
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193 knots		CG 32.4	%MAC
Altitude 12,660 feet		Weight 31,750	lb
Avg. N_2 9150 RPM		Rudder pos. 0	deg.
		Aileron pos. 0	deg.

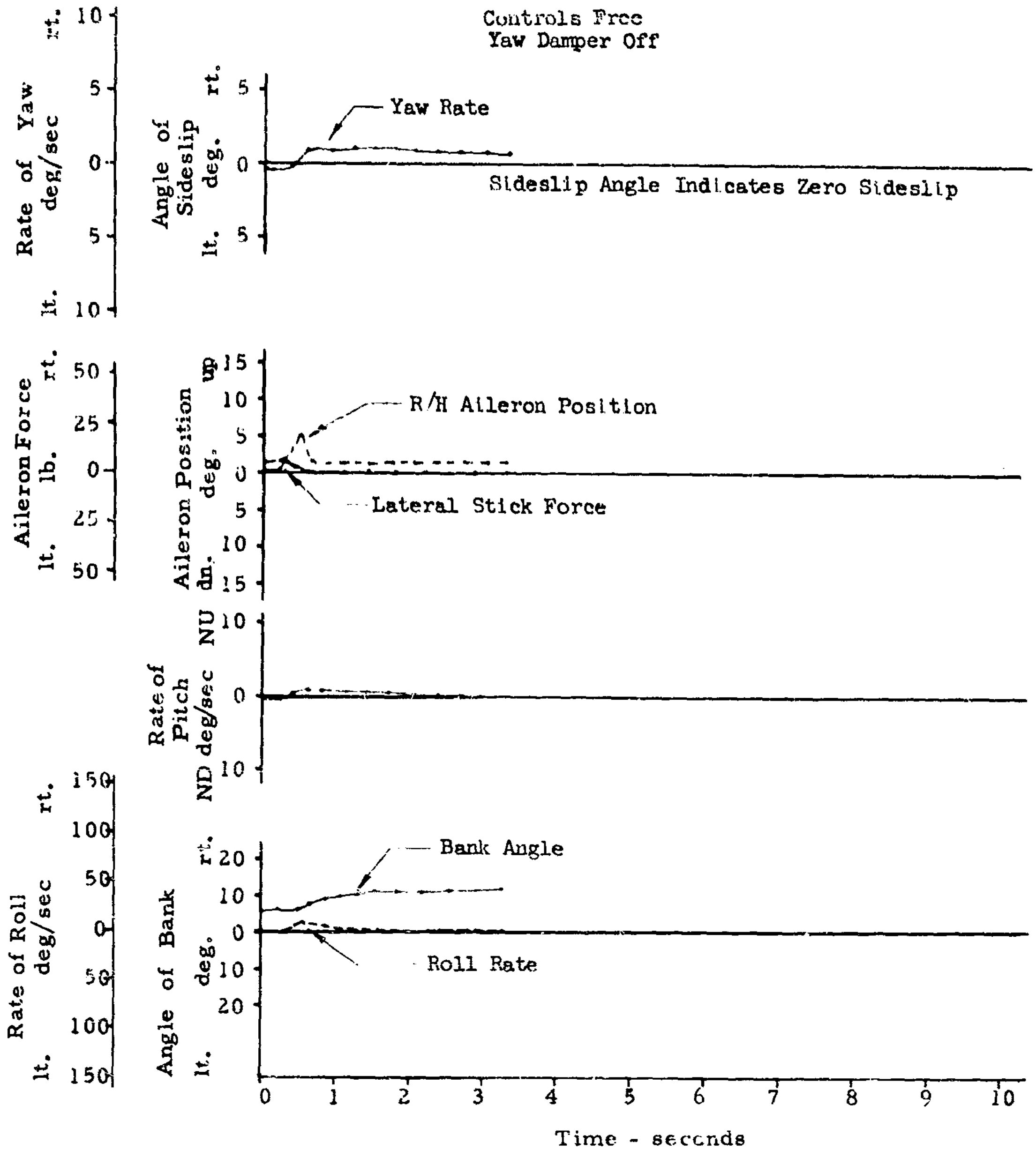

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Figure No. 102
DYNAMIC LATERAL STABILITY
F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS	193 knots	CG	32.4 %MAC
Altitude	12,660 feet	Weight	31,750 lb
Avg. N_2	9150 RPM	Rudder pos.	0 deg.
		Aileron pos.	0 deg.

Controls Free
Yaw Damper Off

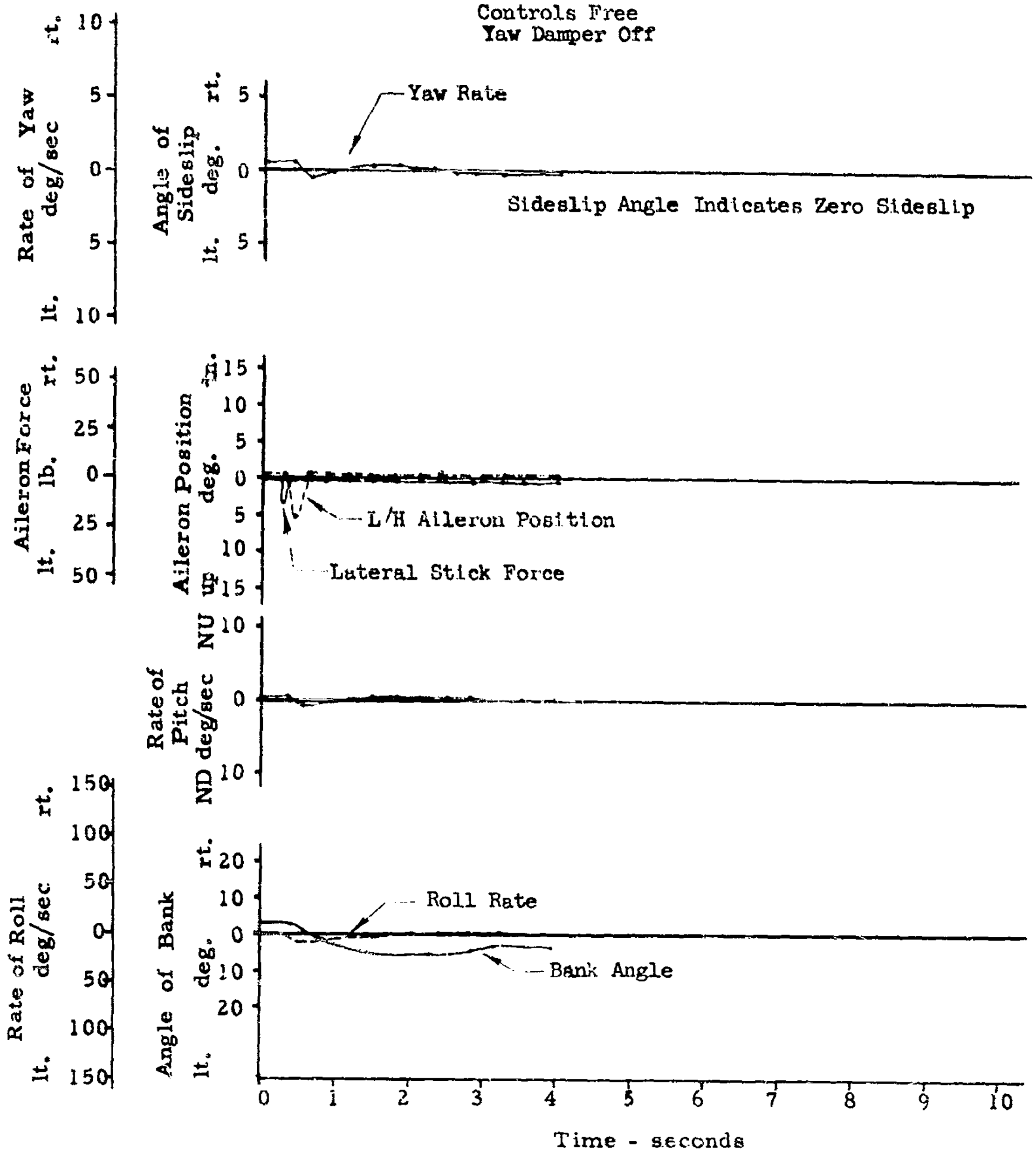

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Figure No. 103

DYNAMIC DIRECTIONAL STABILITY

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

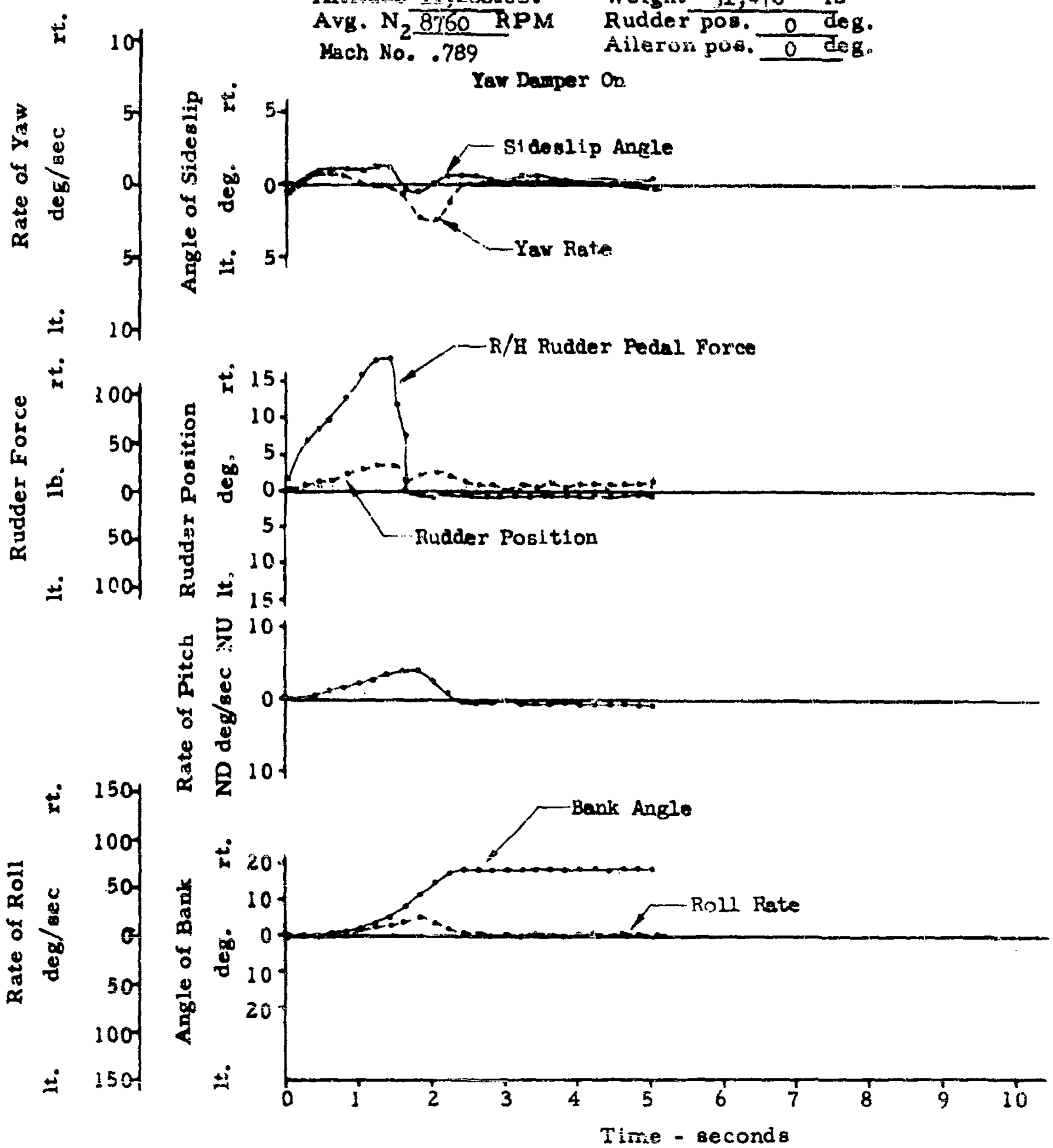
CAS 432 knots — CG 33.9 %MAC

Altitude 11,260 feet — Weight 31,470 lb

Avg. N_2 8760 RPM — Rudder pos. 0 deg.

Mach No. .789 — Aileron pos. 0 deg.

Yaw Damper On



APPENDIX I

Figure No. 104

DYNAMIC DIRECTIONAL STABILITY

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS 432 knots — CG 33.9 %MAC

Altitude 11,260 feet — Weight 31,470 lb

Avg. N_2 8760 RPM — Rudder pos. 0 deg.

Mach No. .789 — Aileron pos. 0 deg.

Yaw Damper On

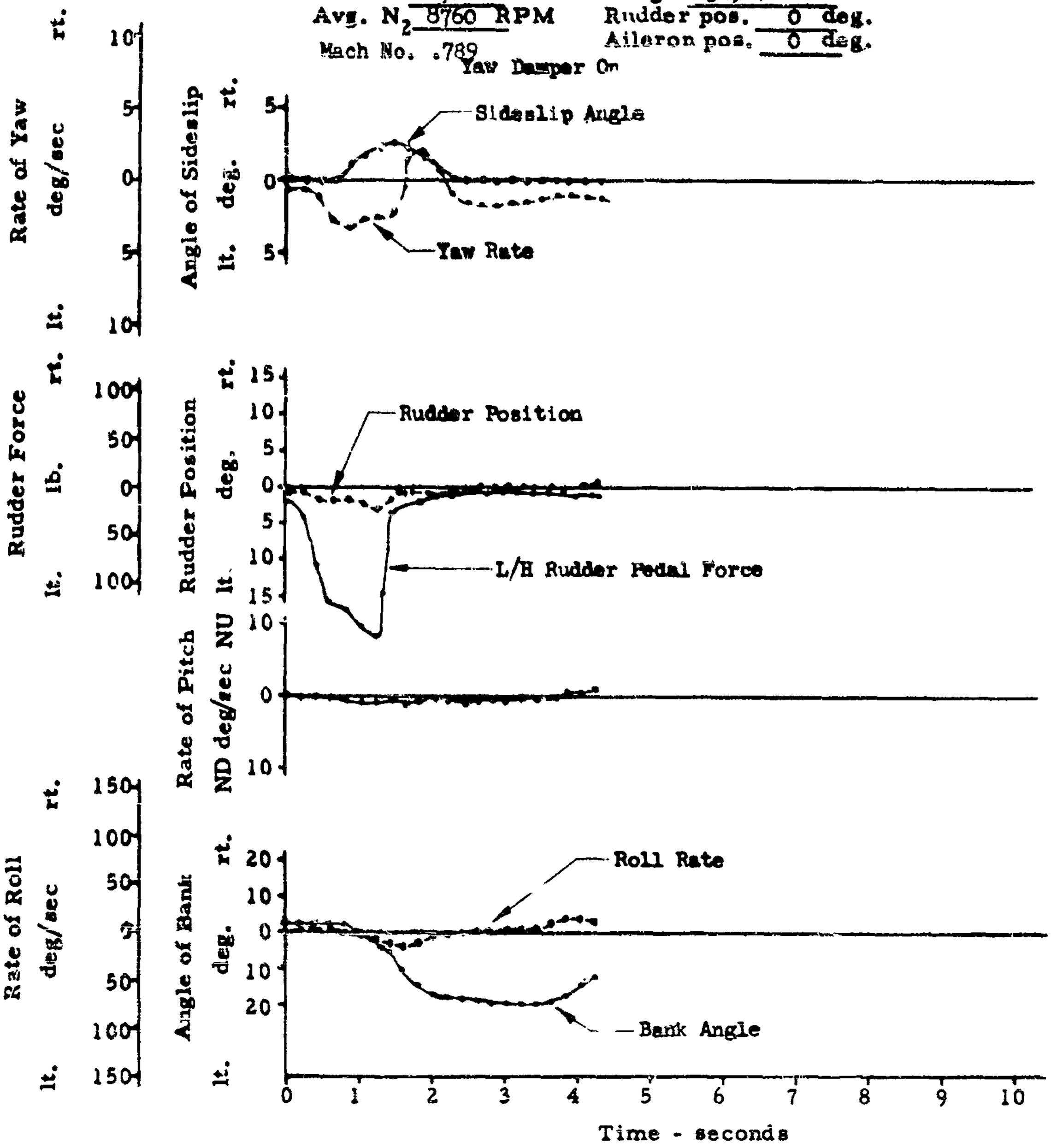


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Figure No. 105

DYNAMIC DIRECTIONAL STABILITY

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

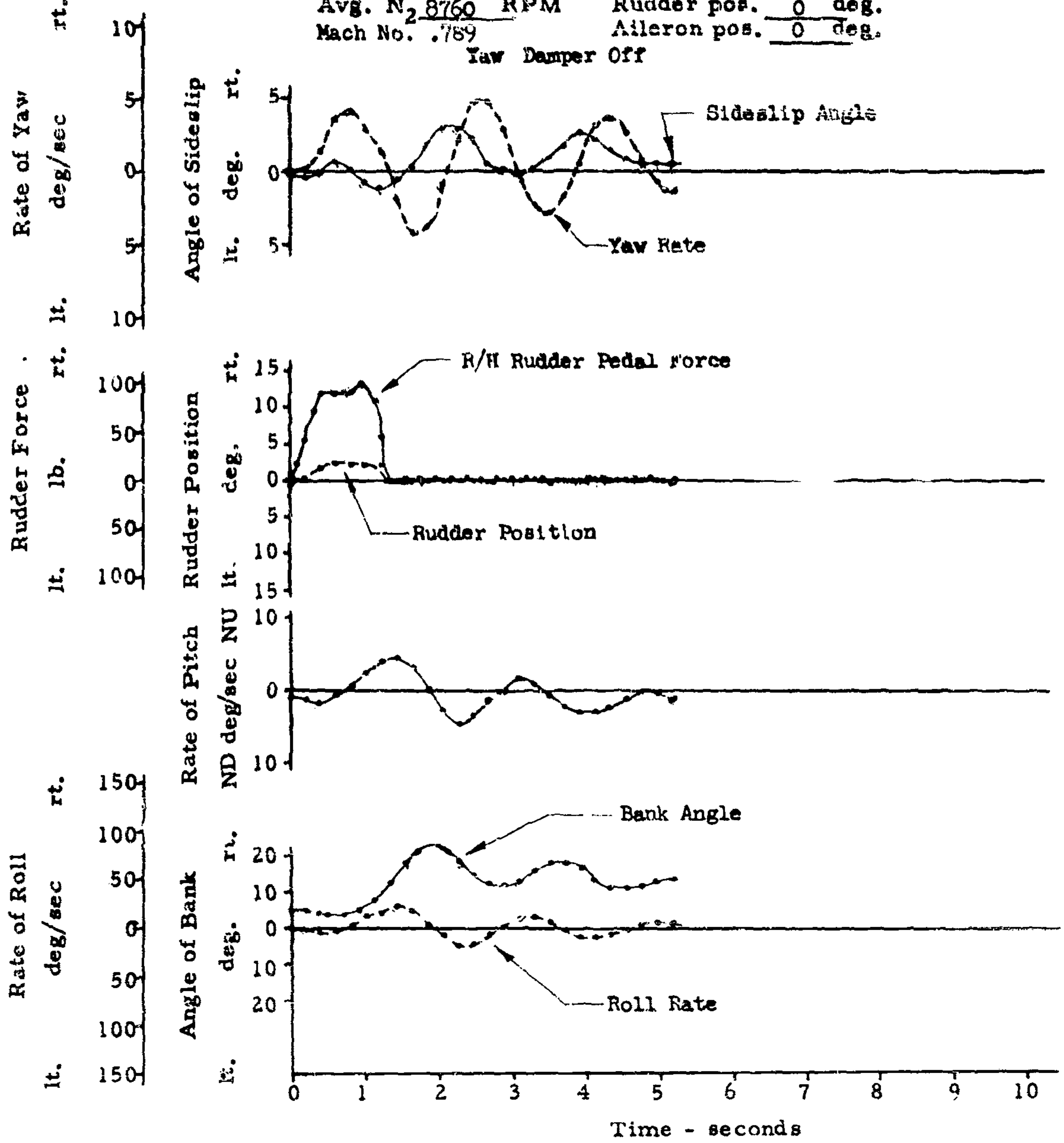
CAS 432 knots CG 33.9 %MAC

Altitude 11,250 feet Weight 31,470 lb

Avg. N_2 8760 RPM Rudder pos. 0 deg.

Mach No. .789 Aileron pos. 0 deg.

Yaw Damper Off



APPENDIX I

Figure No. 106

DYNAMIC DIRECTIONAL STABILITY

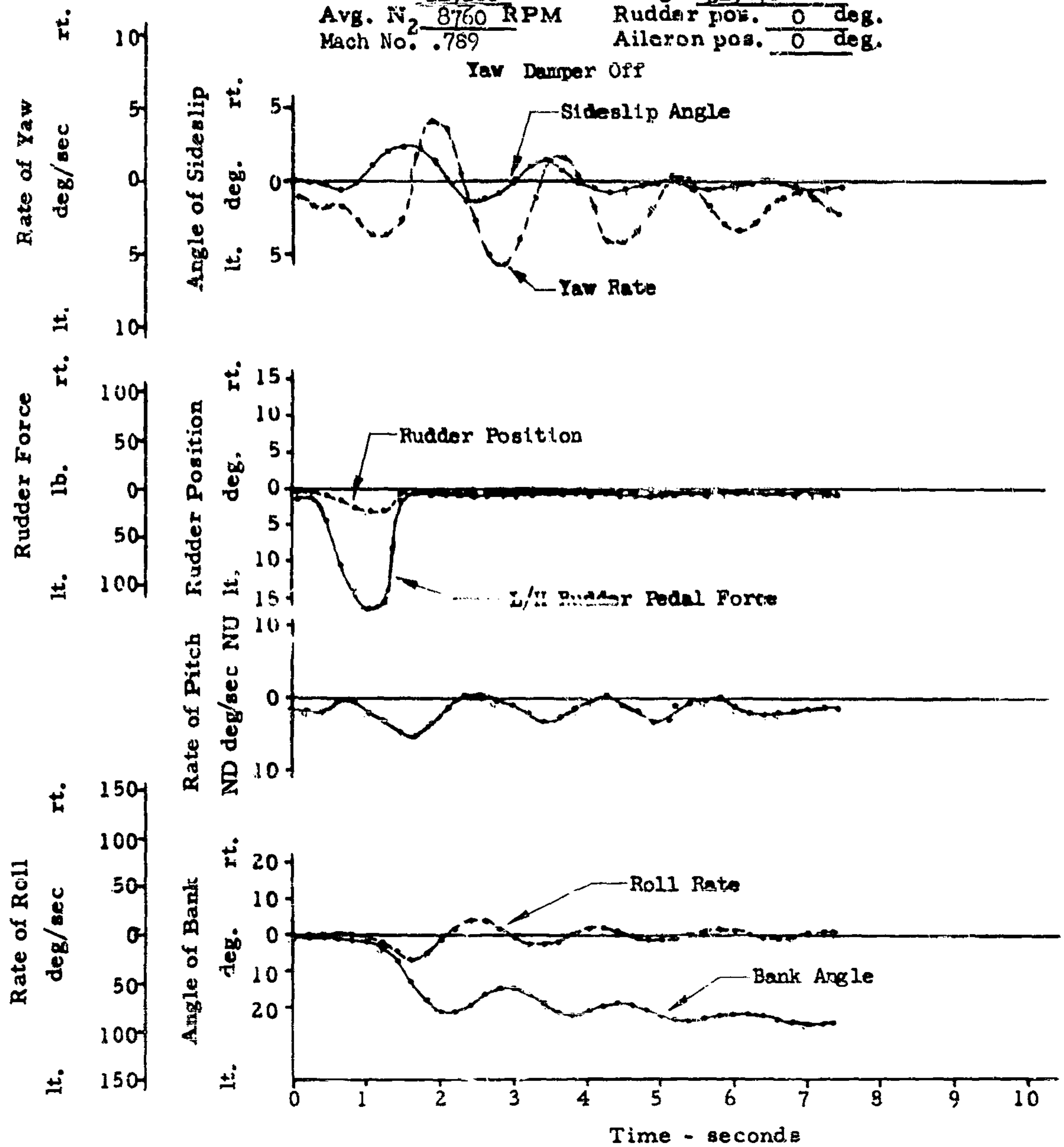
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	432 knots	CG	33.9 %MAC
Altitude	11,260 feet	Weight	31,470 lb
Avg. N_2	8760 RPM	Rudder pos.	0 deg.
Mach No.	.789	Aileron pos.	0 deg.

Yaw Damper Off



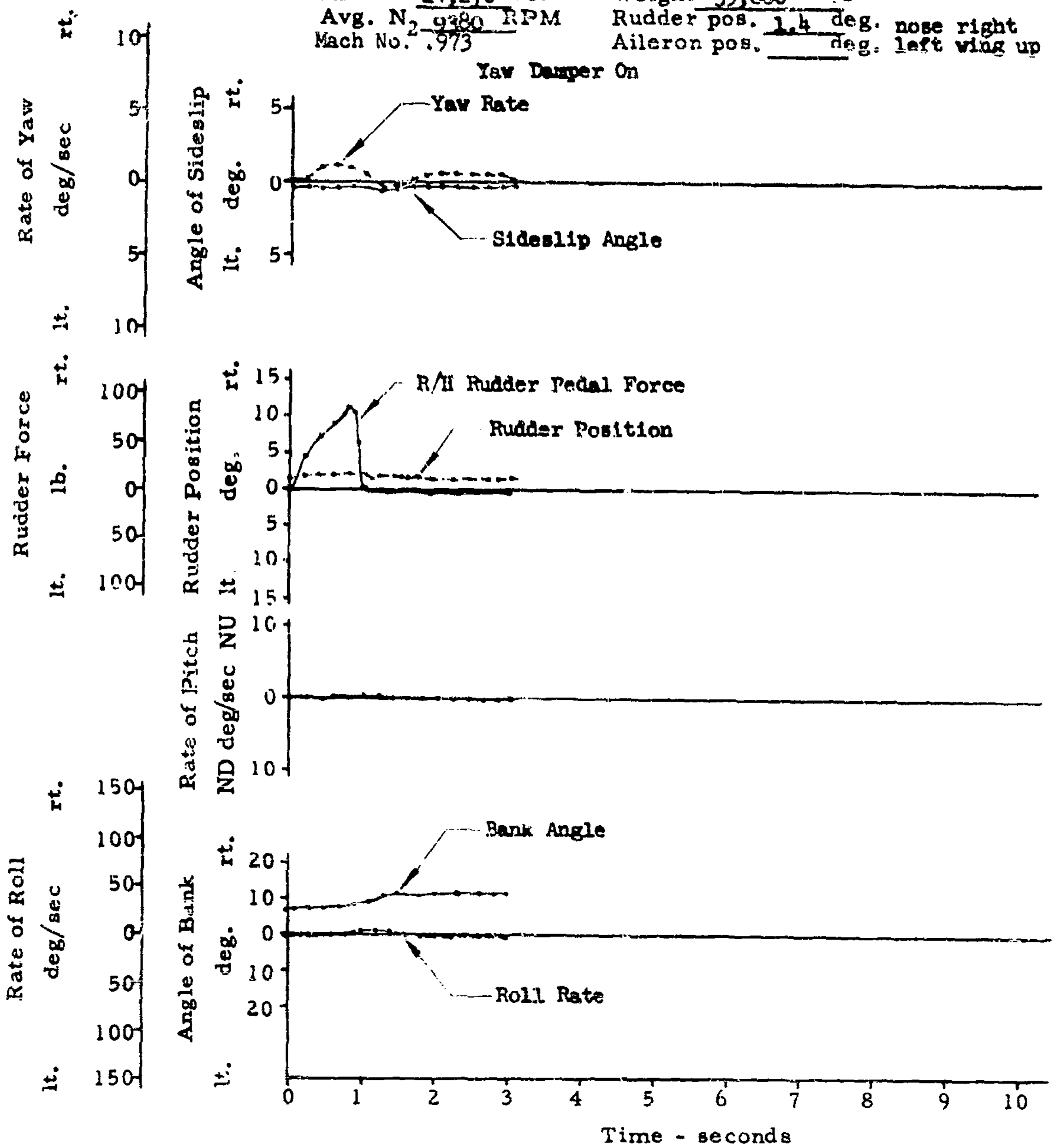
APPENDIX I

Figure No. 107

DYNAMIC DIRECTIONAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots		CG 32.6 %MAC	
Altitude 14,270 feet		Weight 35,860 lb	
Avg. N_2 9380 RPM		Rudder pos. 1.4 deg.	nose right
Mach No. .973		Aileron pos. ___ deg.	left wing up

Yaw Damper On



APPENDIX I

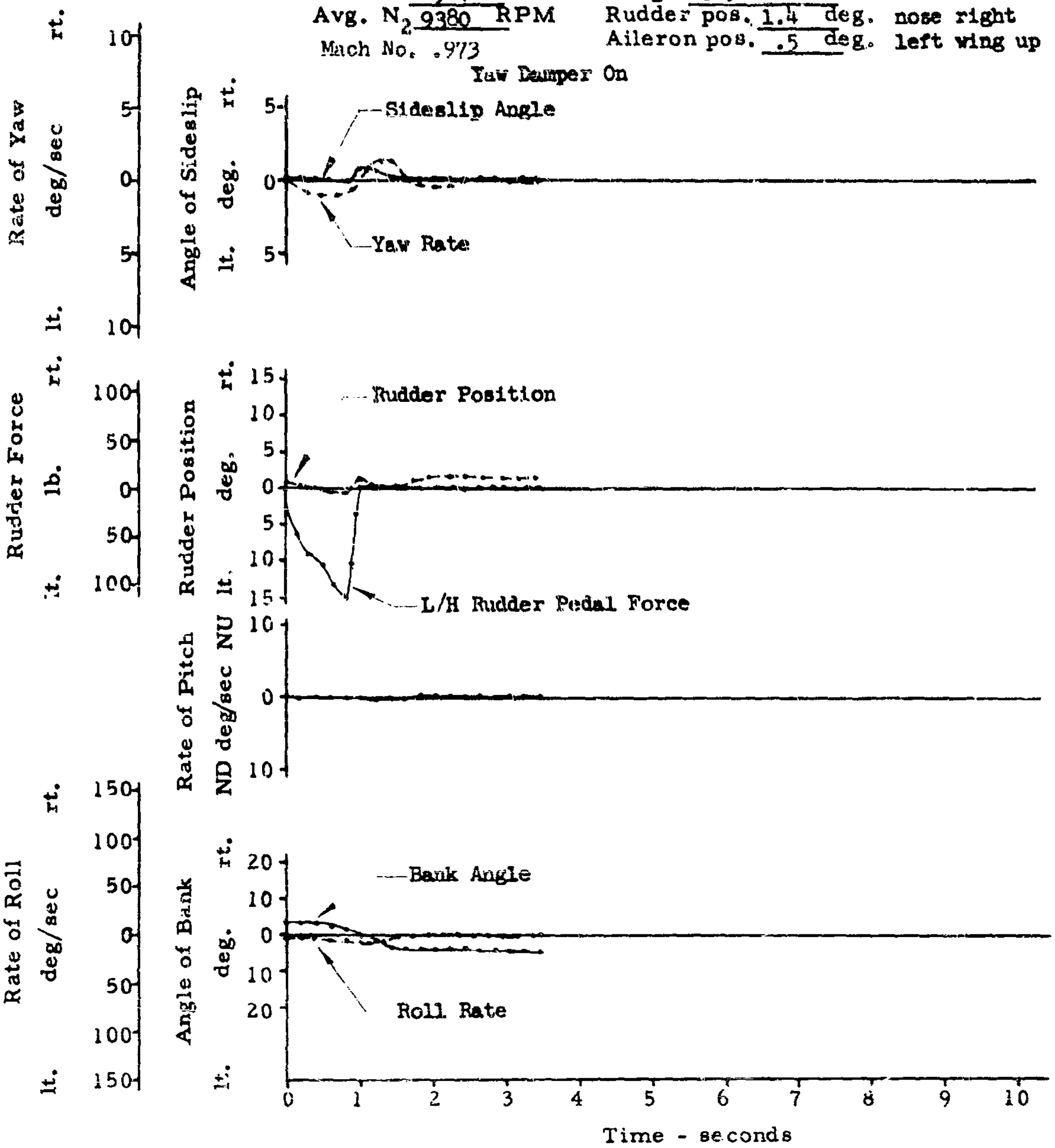
Figure No. 108

DYNAMIC DIRECTIONAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots
Altitude 14,270 feet
Avg. N_2 9380 RPM
Mach No. .973

CG 32.6 %MAC
Weight 35,860 lb
Rudder pos. 1.4 deg. nose right
Aileron pos. .5 deg. left wing up

Yaw Damper On



APPENDIX I

Figure No. 109

DYNAMIC DIRECTIONAL STABILITY

F-101A, USAF No. 53-2419

Combat Configuration

TRIM CONDITIONS

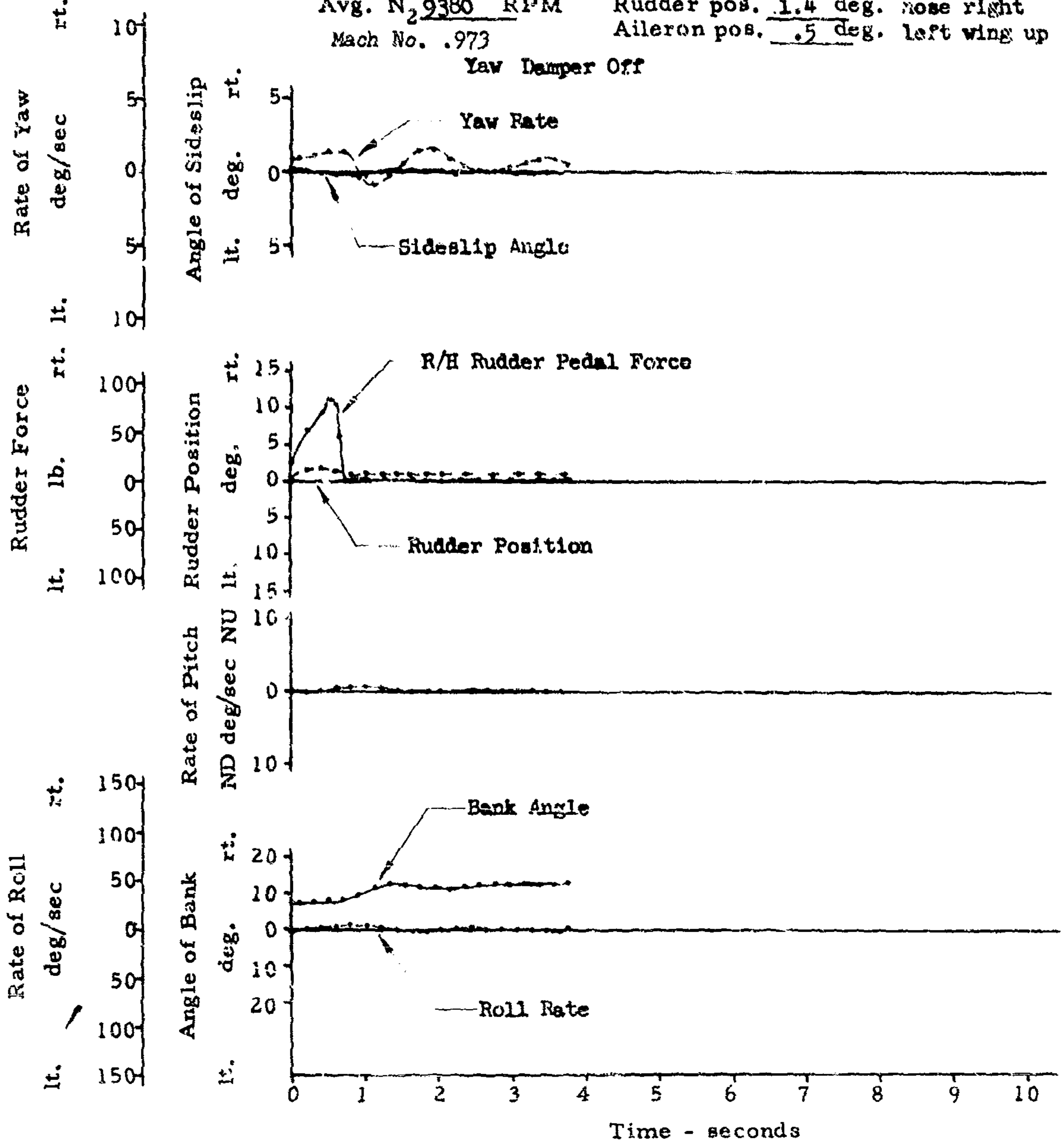
CAS 51. knots CG 32.6 %MAC

Altitude 14,270 feet Weight 35,860 lb

Avg. N_2 9380 RPM Rudder pos. 1.4 deg. nose right

Mach No. .973 Aileron pos. .5 deg. left wing up

Yaw Damper Off

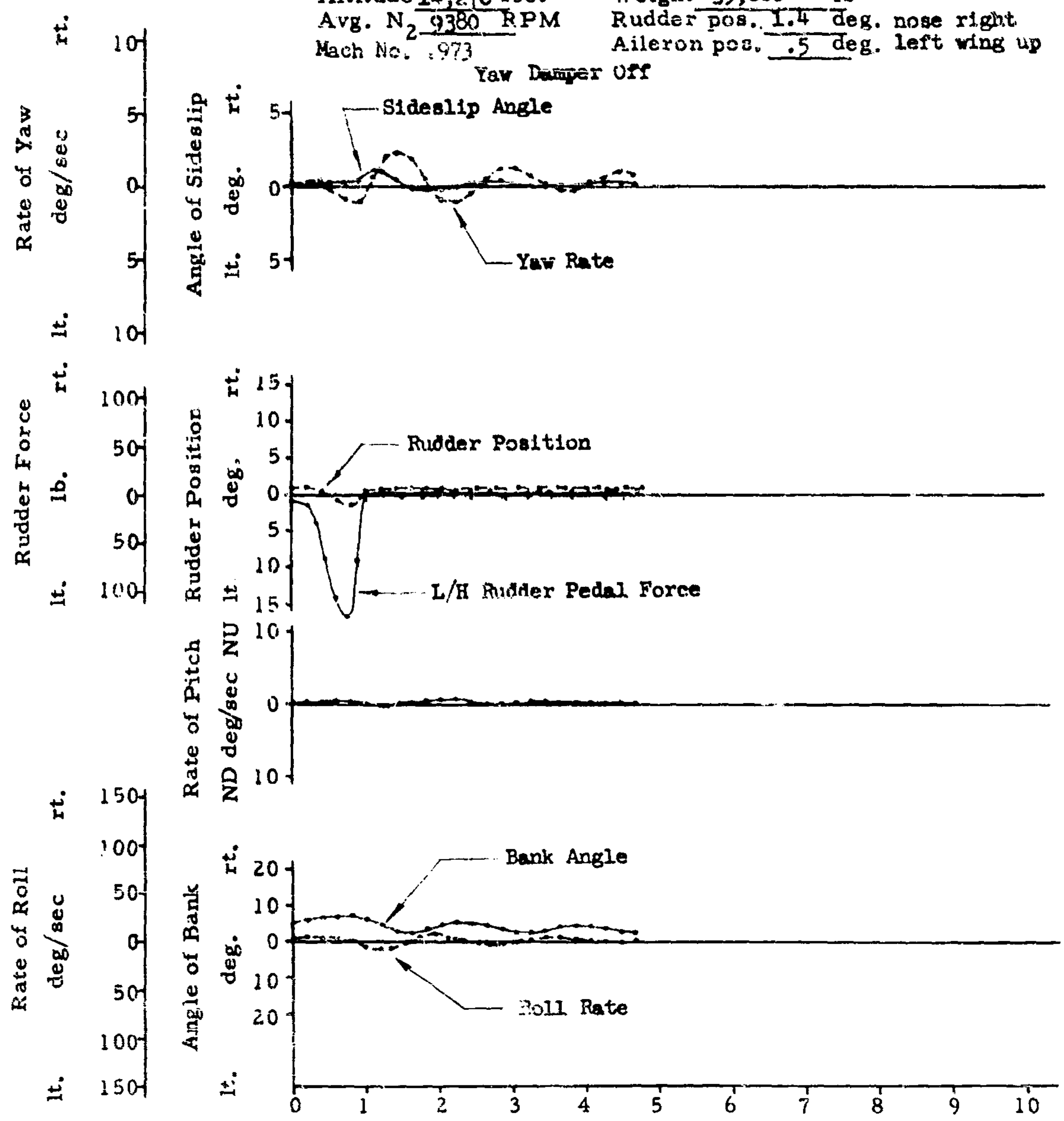


APPENDIX I

Figure No. 110

DYNAMIC DIRECTIONAL STABILITY
F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS 511 knots CG 32.6 %MAC
Altitude 14,270 feet Weight 35,860 lb
Avg. N_2 9380 RPM Rudder pos. 1.4 deg. nose right
Mach No. .973 Aileron pos. .5 deg. left wing up
Yaw Damper Off



APPENDIX I

Figure No. 111

DYNAMIC DIRECTIONAL STABILITY

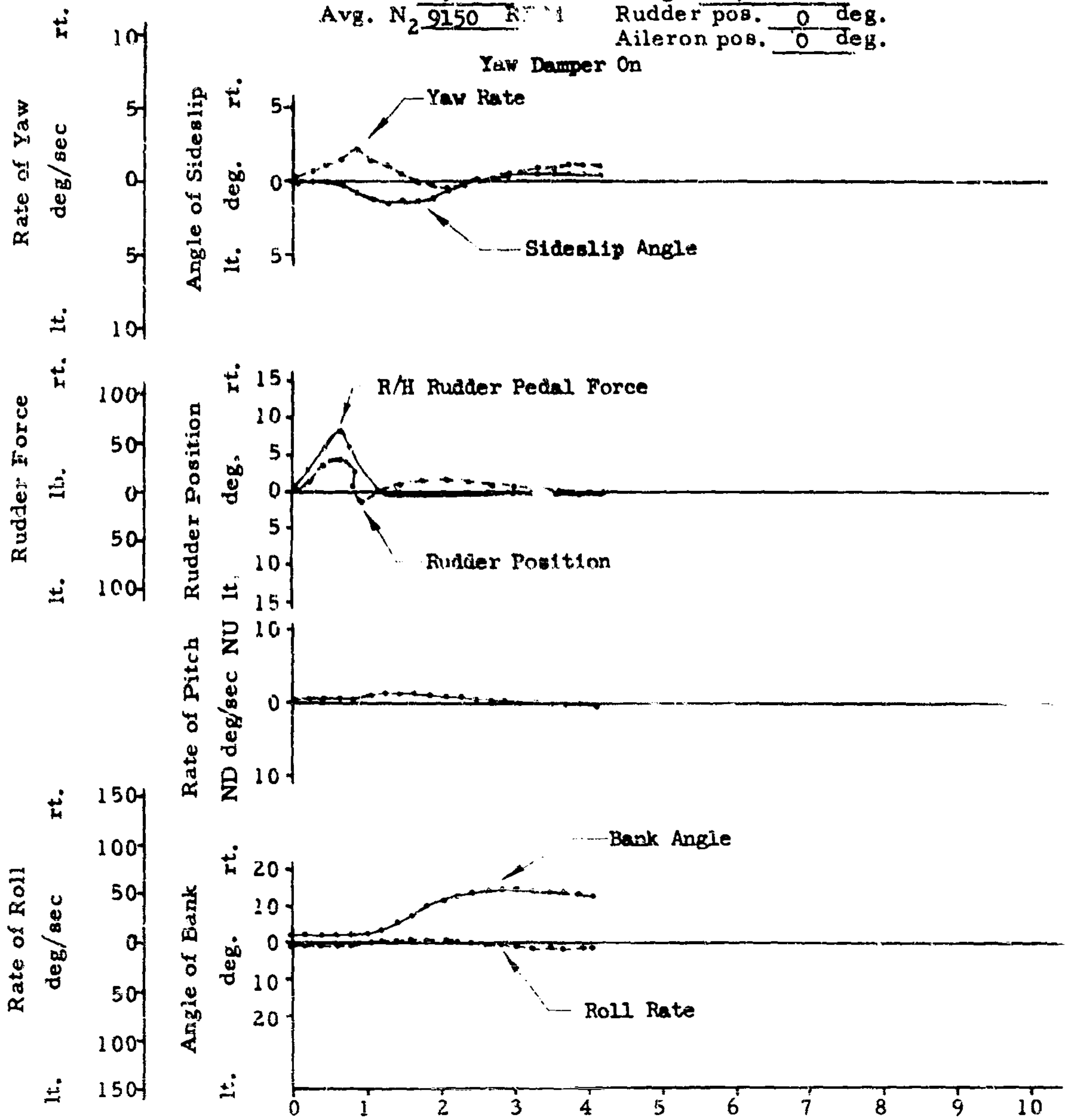
F-101A, USAF No. 53-2419

Power Approach Configuration

TRIM CONDITIONS

CAS	193 knots	CG	32.4 %MAC
Altitude	12,660 feet	Weight	31,750 lb
Avg. N_2	9150 RPM	Rudder pos.	0 deg.
		Aileron pos.	0 deg.

Yaw Damper On



APPENDIX I

Figure No. 112

DYNAMIC DIRECTIONAL STABILITY

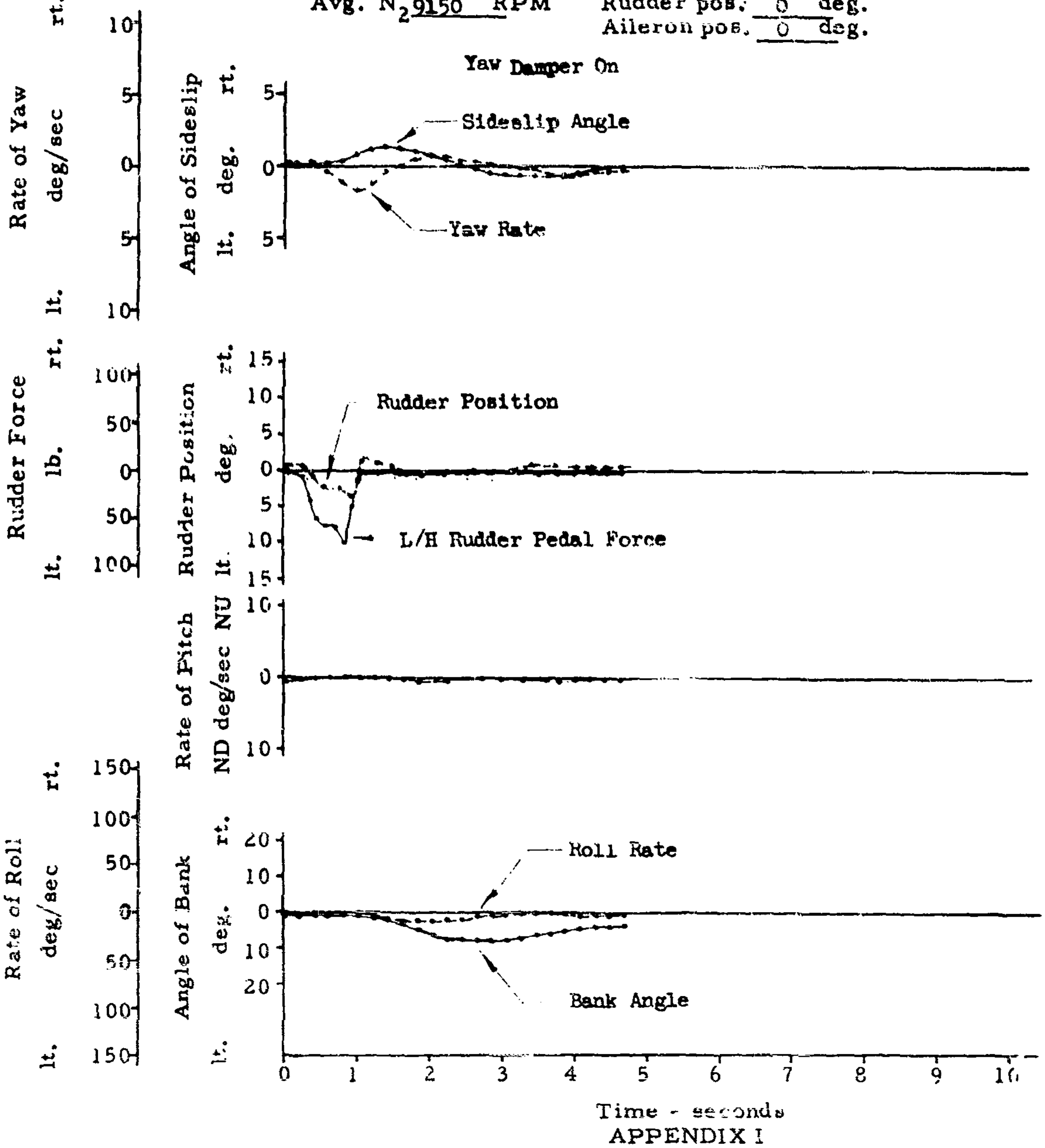
F-101A, USAF No. 53-2419

Power Approach Configuration

TRIM CONDITIONS

CAS 193 knots		CG 32.4 %MAC	
Altitude 12,660 feet		Weight 31,750 lb	
Avg. N_2 9150 RPM		Rudder pos. 0 deg.	
		Aileron pos. 0 deg.	

Yaw Damper On



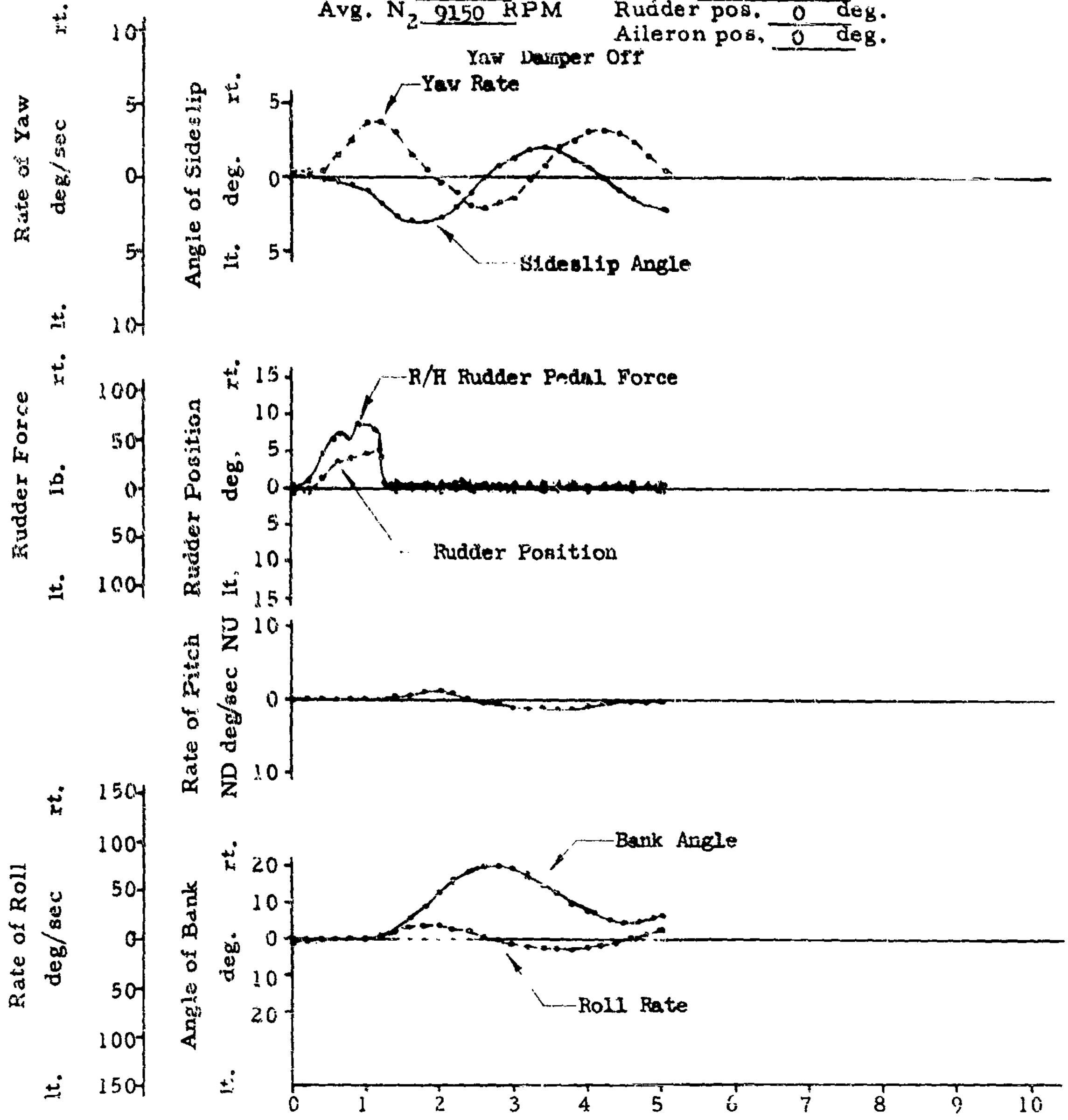
APPENDIX I

Figure No. 113

DYNAMIC DIRECTIONAL STABILITY
F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193 knots		CG 32.4	%MAC
Altitude 12,660 feet		Weight 31,750	lb
Avg. N_2 9150 RPM		Rudder pos. 0	deg.
		Aileron pos. 0	deg.

Yaw Damper Off



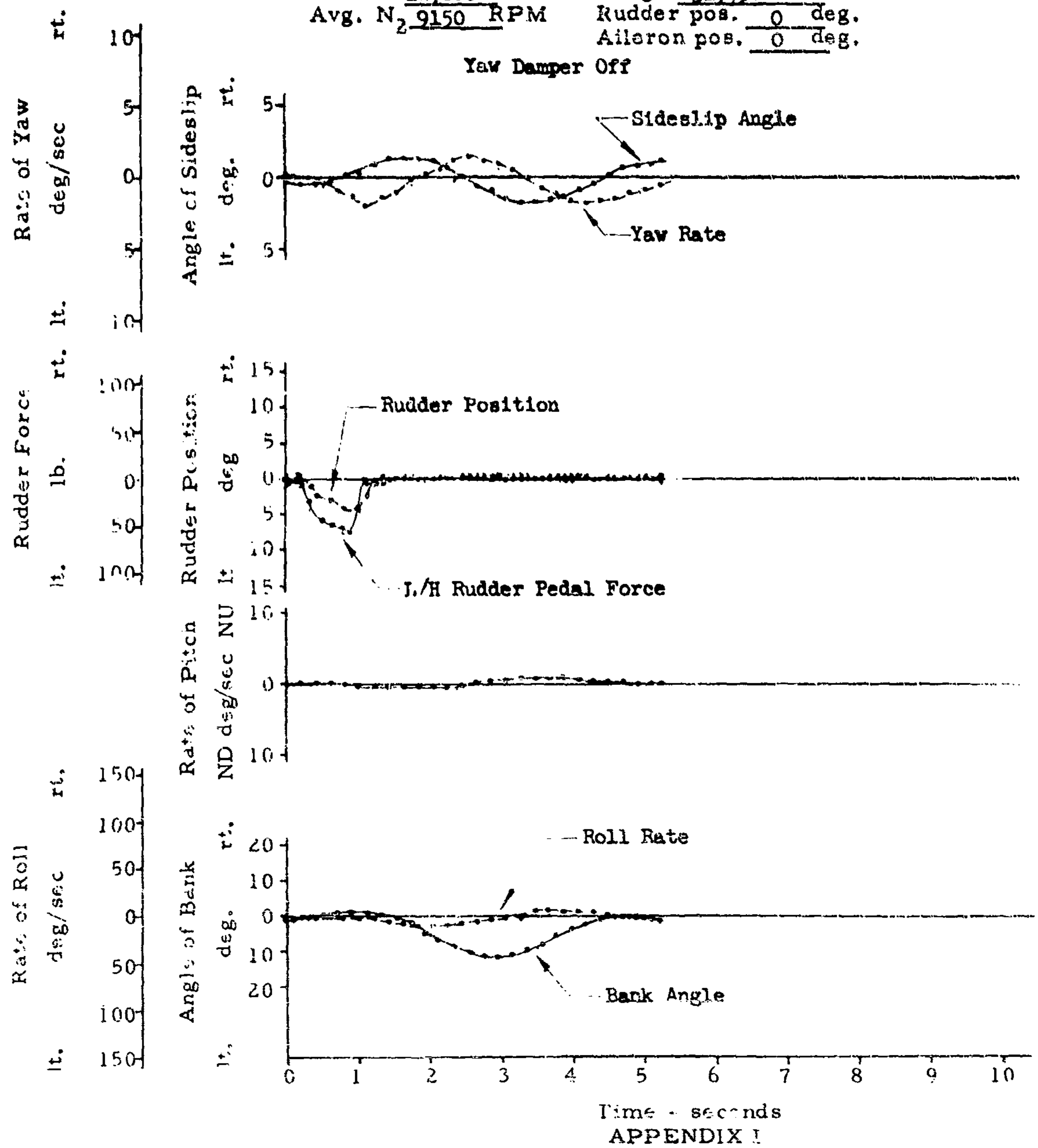
APPENDIX I

Figure No. 114

DYNAMIC DIRECTIONAL STABILITY
F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS 193 knots		CG 32.4 %MAC	
Altitude 12,660 feet		Weight 31,750 lb	
Avg. N_2 9150 RPM		Rudder pos. 0 deg.	
		Aileron pos. 0 deg.	

Yaw Damper Off



APPENDIX I

Figure No. 115

DYNAMIC DIRECTIONAL STABILITY

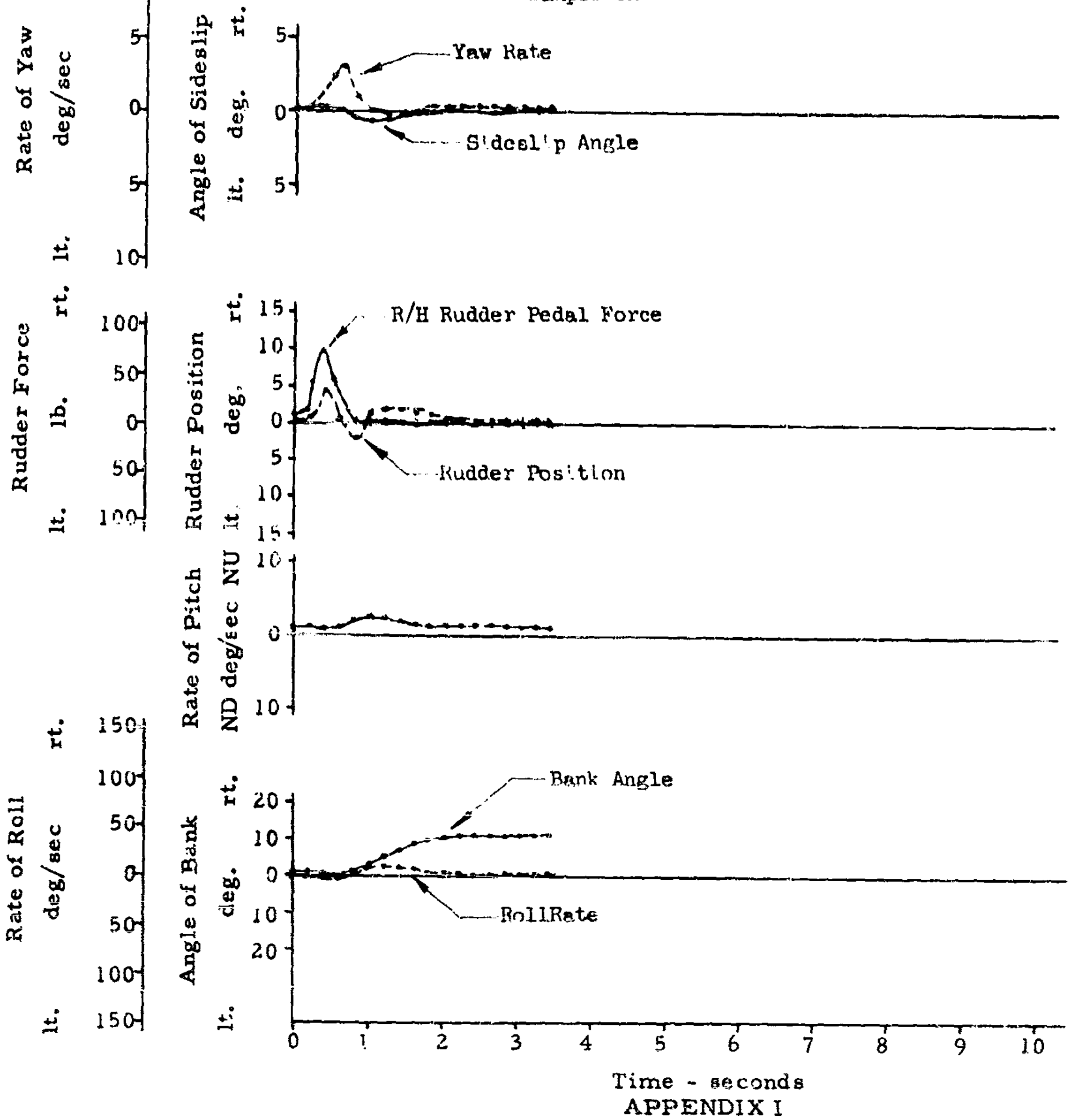
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	270.5 knots	CG	33.6 %MAC
Altitude	38,070 feet	Weight	32,430 lb
Avg. N_2	8280 RPM	Rudder pos.	0 deg.
Mach No.	.848	Aileron pos.	0 deg.

Yaw Damper On



APPENDIX I

Figure No. 116

DYNAMIC DIRECTIONAL STABILITY

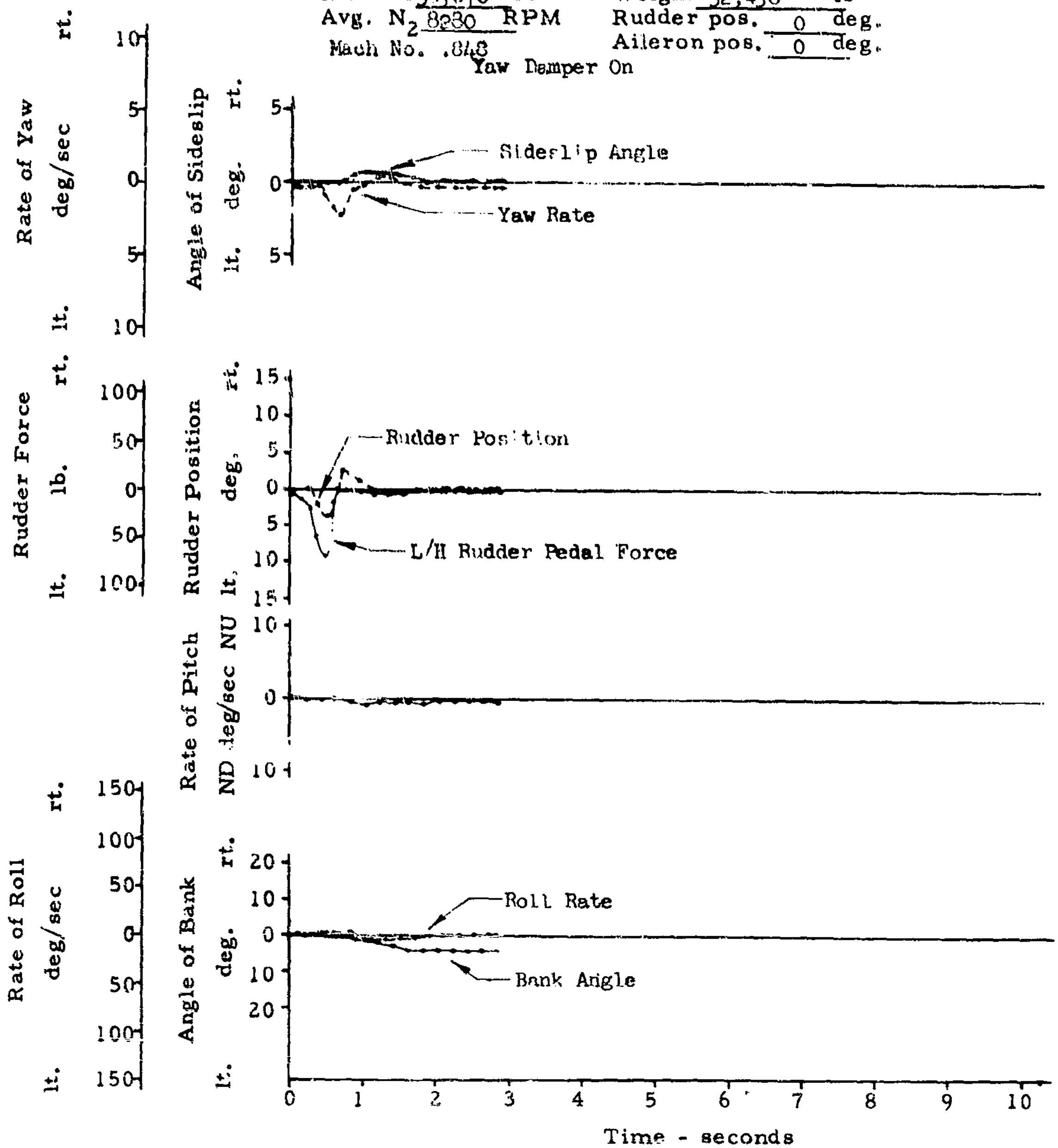
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS 270.5 knots		CG 33.6	%MAC
Altitude 38,070 feet		Weight 32,430	lb
Avg. N_2 8280 RPM		Rudder pos. 0	deg.
Mach No. .848		Aileron pos. 0	deg.

Yaw Damper On



APPENDIX I

Figure No. 117

DYNAMIC DIRECTIONAL STABILITY

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

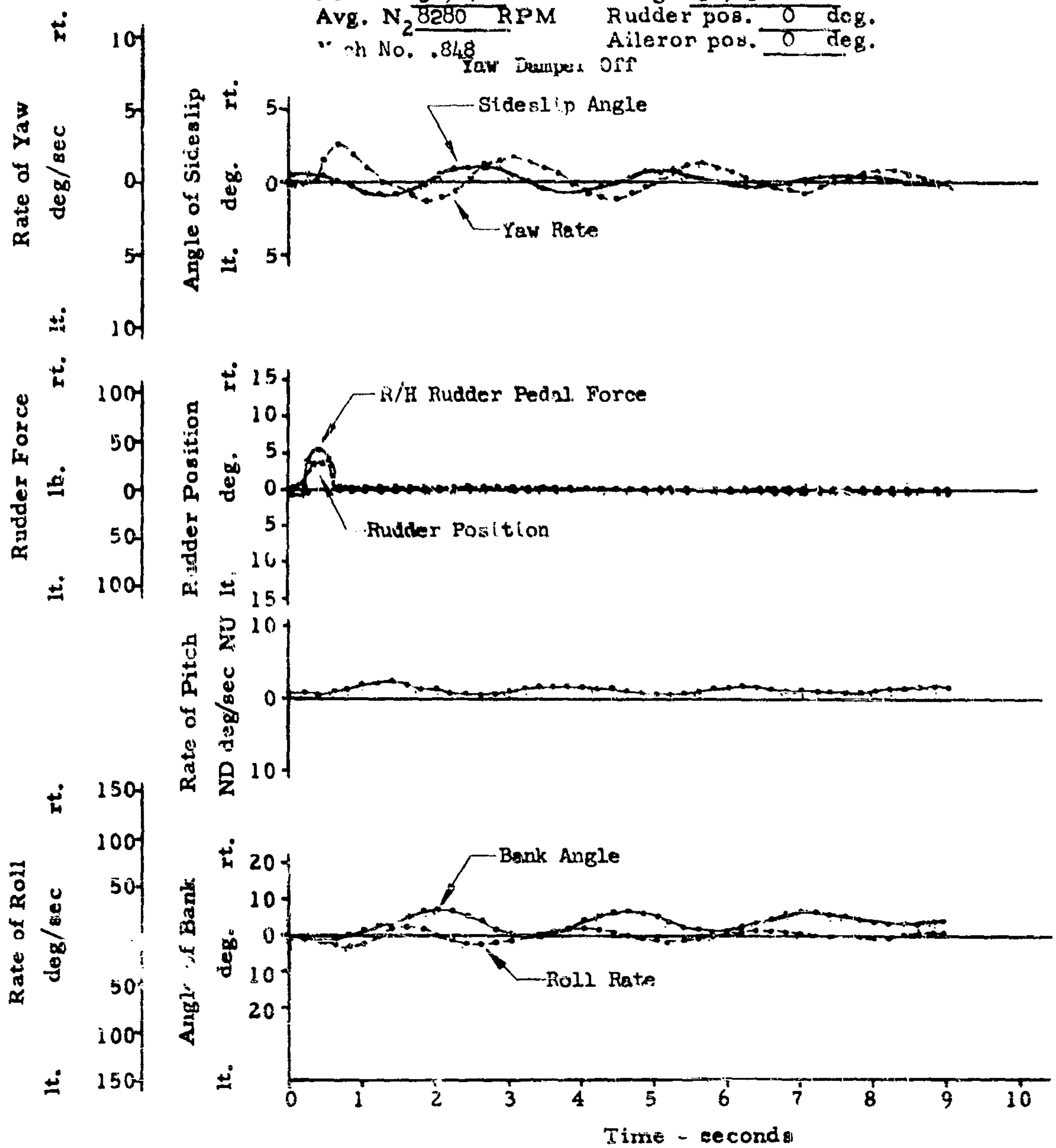
CAS 270.5 knots CG 33.6 %MAC

Altitude 38,070 feet Weight 32,430 lb

Avg. N_2 8280 RPM Rudder pos. 0 deg.

Mach No. .848 Aileron pos. 0 deg.

Yaw Damper Off



APPENDIX I

Figure No. 118

DYNAMIC DIRECTIONAL STABILITY

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

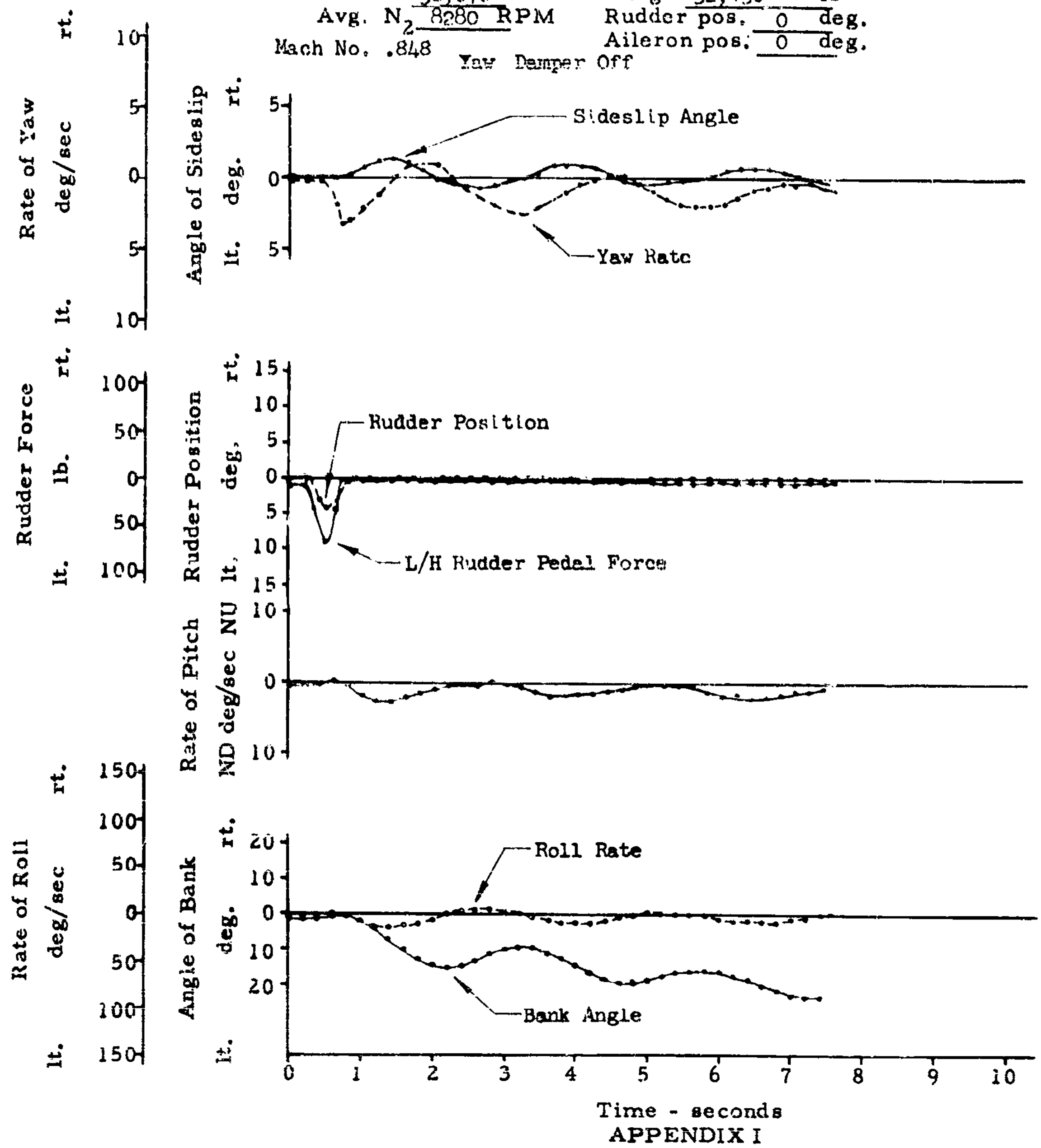
CAS 270.5 knots CG 33.6 %MAC

Altitude 38,070 feet Weight 32,430 lb

Avg. N_2 8280 RPM Rudder pos. 0 deg.

Mach No. .848 Aileron pos. 0 deg.

Yaw Damper Off



APPENDIX I

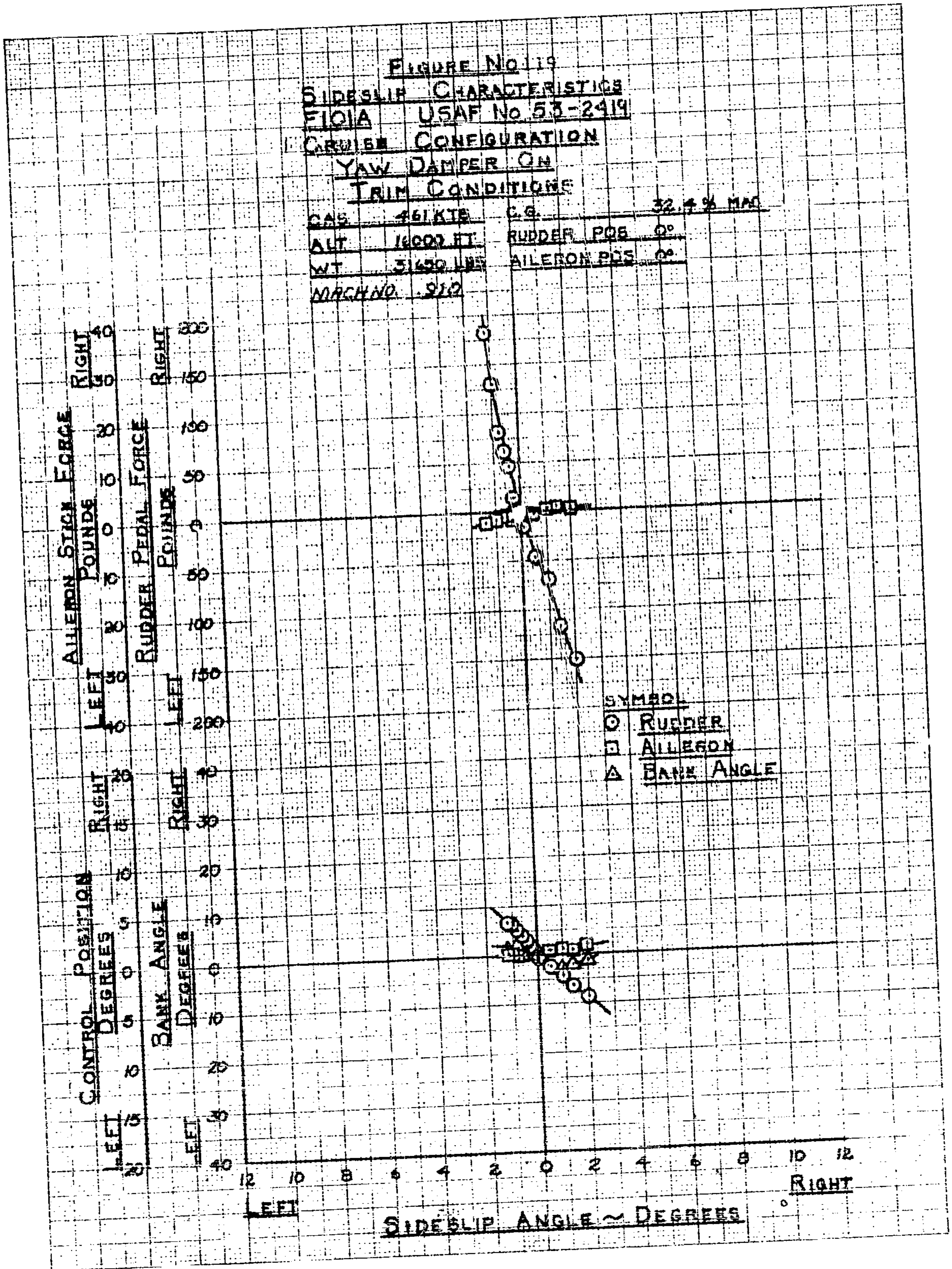


FIGURE NO. 119
SIDESLIP CHARACTERISTICS
F101A USAF No. 53-2419
CRUISE CONFIGURATION
YAW DAMPER ON
TRIM CONDITIONS
CAS 461 KTS
ALT 16000 FT
WT 31450 LBS
MACH NO. .910
C.G. 32.4 % MAC
RUDDER POS 0°
AILERON POS 0°
AILERON STICK FORCE POUNDS
RIGHT
LEFT
RUDDER PEDAL FORCE POUNDS
CONTROL POSITION DEGREES
BANK ANGLE DEGREES
SYMBOL
RUDDER
AILERON
BANK ANGLE
LEFT
RIGHT
SIDESLIP ANGLE ~ DEGREES

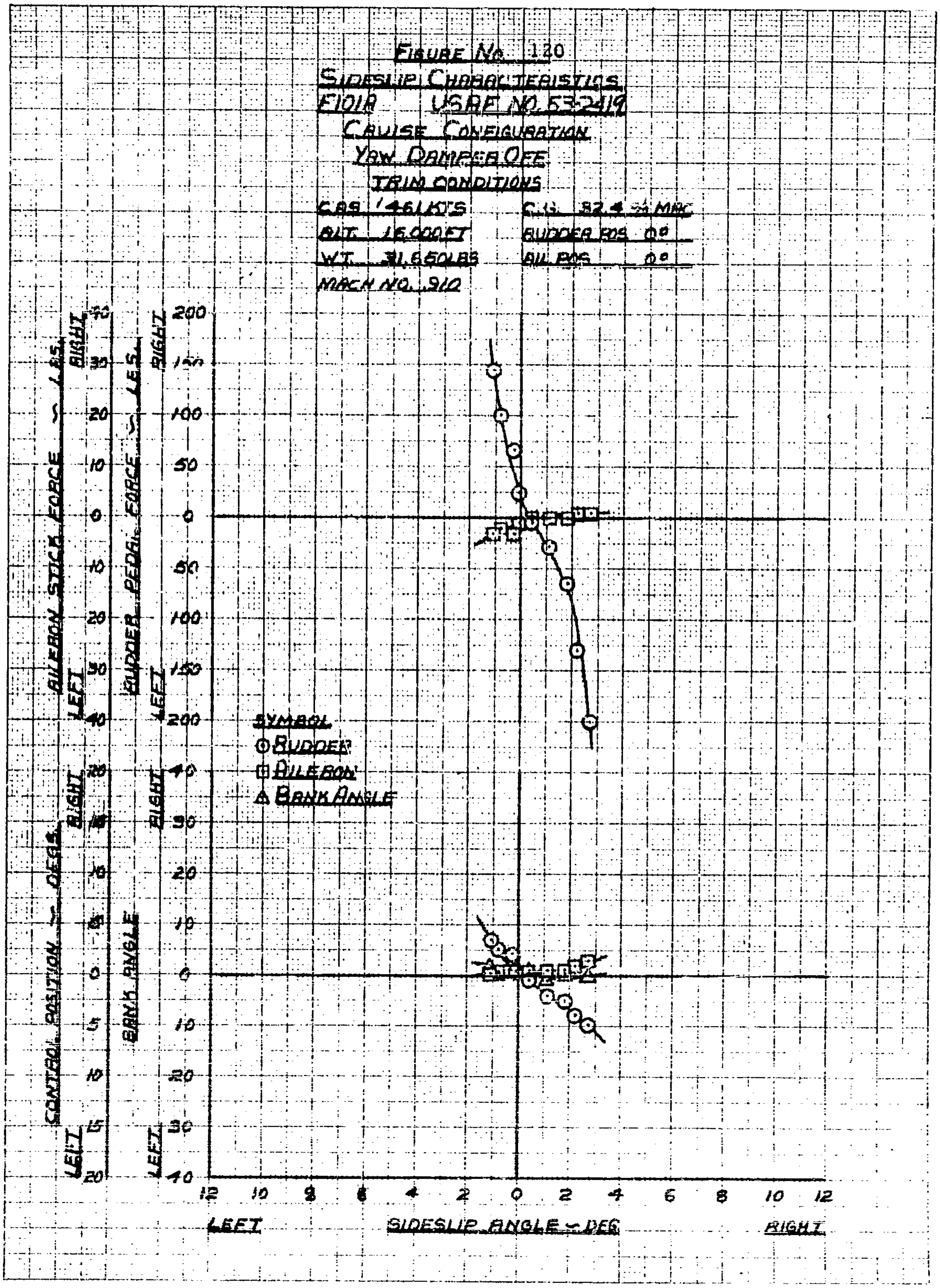


FIGURE No. 120
SIDESLIP CHARACTERISTICS
F101A USAF NO. 53-2419
CRUISE CONFIGURATION
YAW DAMPER OFF
TRIM CONDITIONS
CAS 461 KTS
ALT. 16,000 FT
WT. 31,650 LBS
MACH NO. .910
C.G. 32.4 % MAC
RUDDER POS. 0°
AIL POS. 0°
AILERON STICK FORCE ~ LBS.
RUDDER PEDAL FORCE ~ LBS.
CONTROL POSITION ~ DEGS.
BANK ANGLE
SYMBOL
RUDDER
AILERON
BANK ANGLE
LEFT
SIDESLIP ANGLE ~ DEG
RIGHT

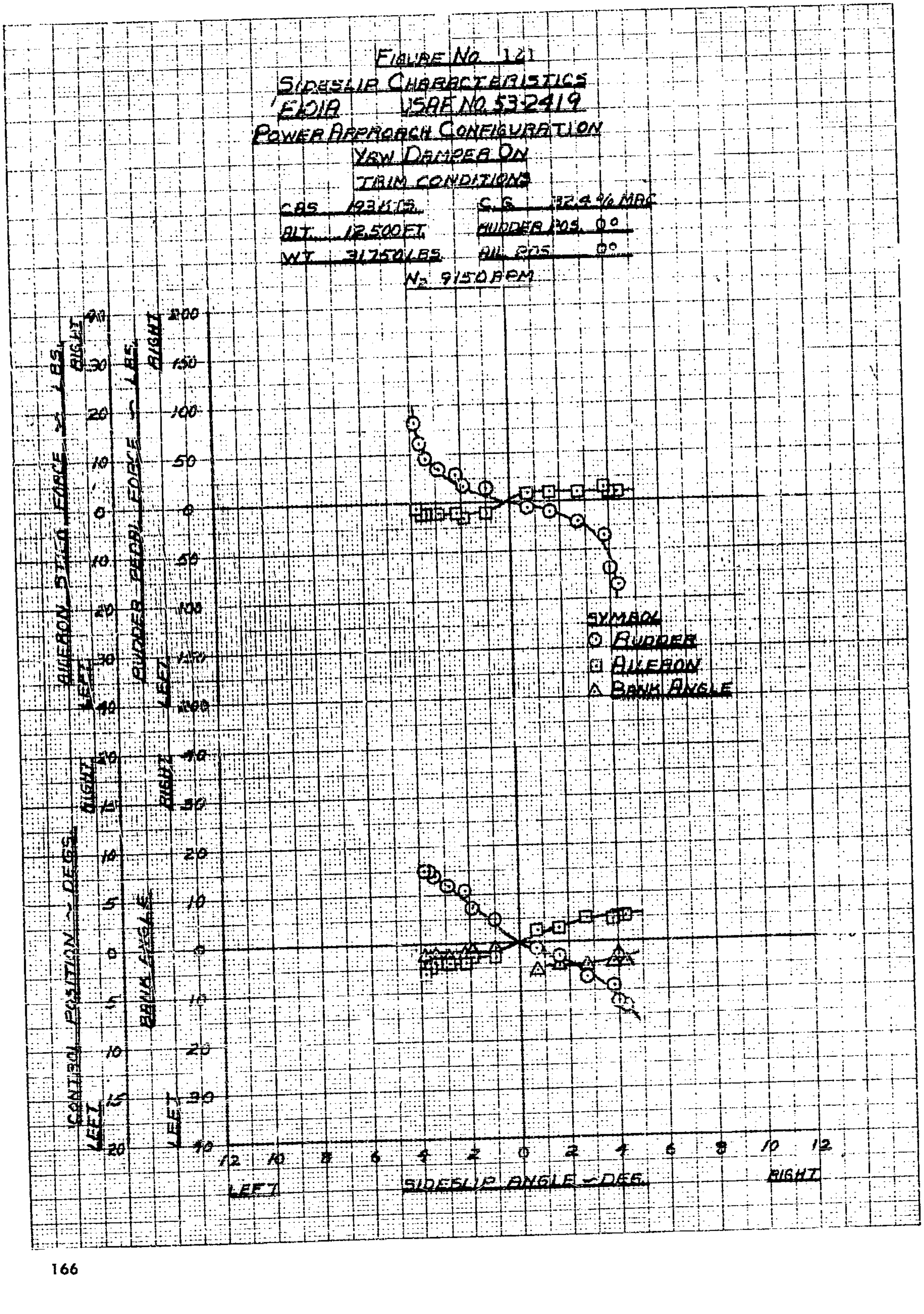


FIGURE No. 121
SIDESLIP CHARACTERISTICS
F101A USAF No. 53-2419
POWER APPROACH CONFIGURATION
YAW DAMPER ON
TRIM CONDITIONS
CAS 193 KTS.
C.G. 32.4 % MAC
ALT. 12,500 FT.
RUDDER POS. 0°
WT. 31750 LBS.
AIL. POS. 0°
N2 9150 RPM
AILERON STICK FORCE ~ LBS.
RUDDER PEDAL FORCE ~ LBS.
RIGHT
LEFT
CONTROL POSITION ~ DEGS.
BANK ANGLE
SYMBOL
RUDDER
AILERON
BANK ANGLE
SIDESLIP ANGLE ~ DEG.
LEFT
RIGHT

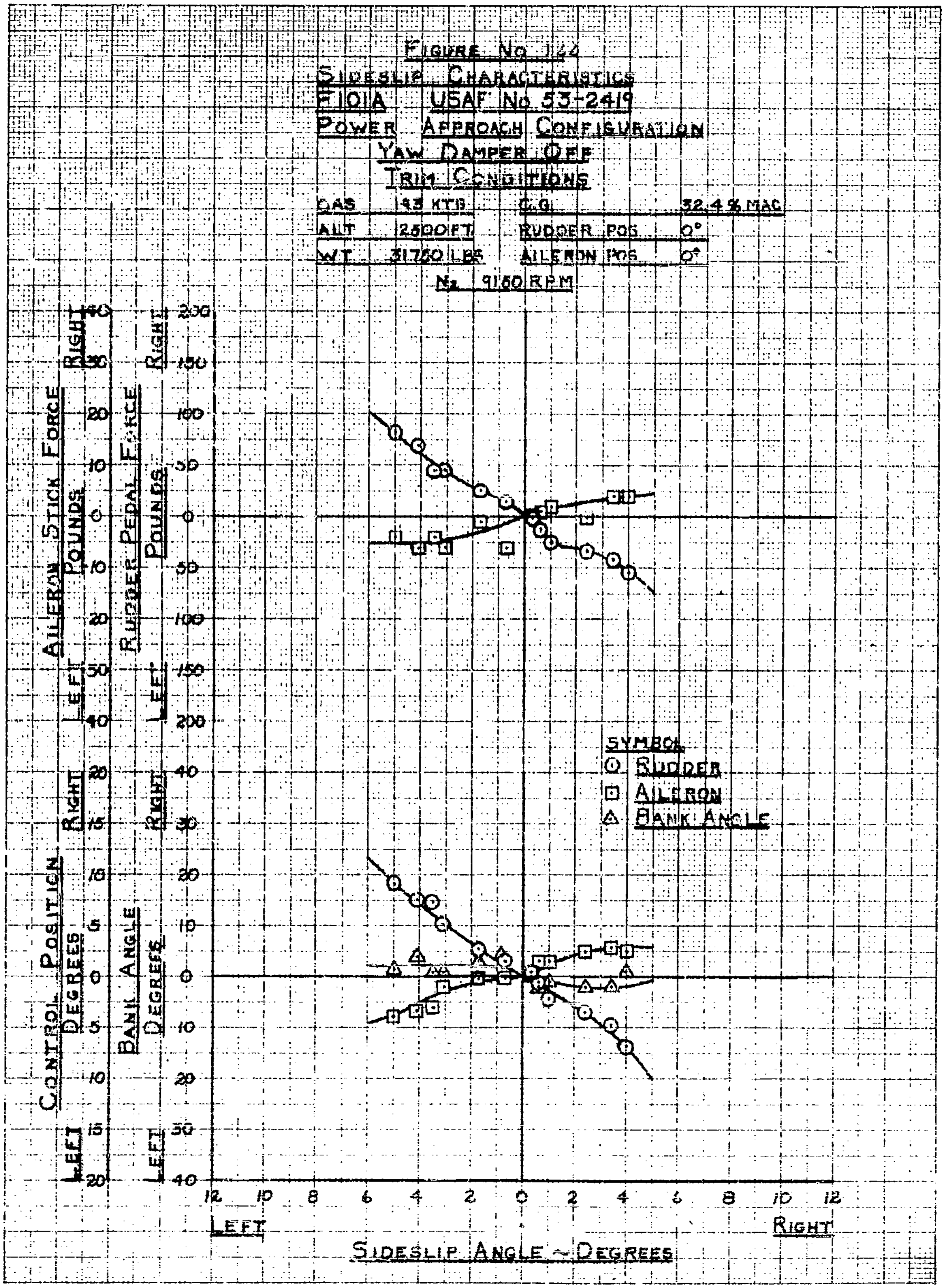


Figure No. 144
Sideslip Characteristics
F101A USAF No. 53-2419
Power Approach Configuration
Yaw Damper Off
Trim Conditions
CAS 143 KTS C.G. 32.4 % MAC
ALT 2500 FT RUDDER POS 0°
WT 31750 LBS AILERON POS 0°
N_2 9150 RPM
Aileron Stick Force Pounds
Rudder Pedal Force Pounds
Control Position Degrees
Bank Angle Degrees
Right
Left
Symbol
Rudder
Aileron
Bank Angle
Sideslip Angle ~ Degrees

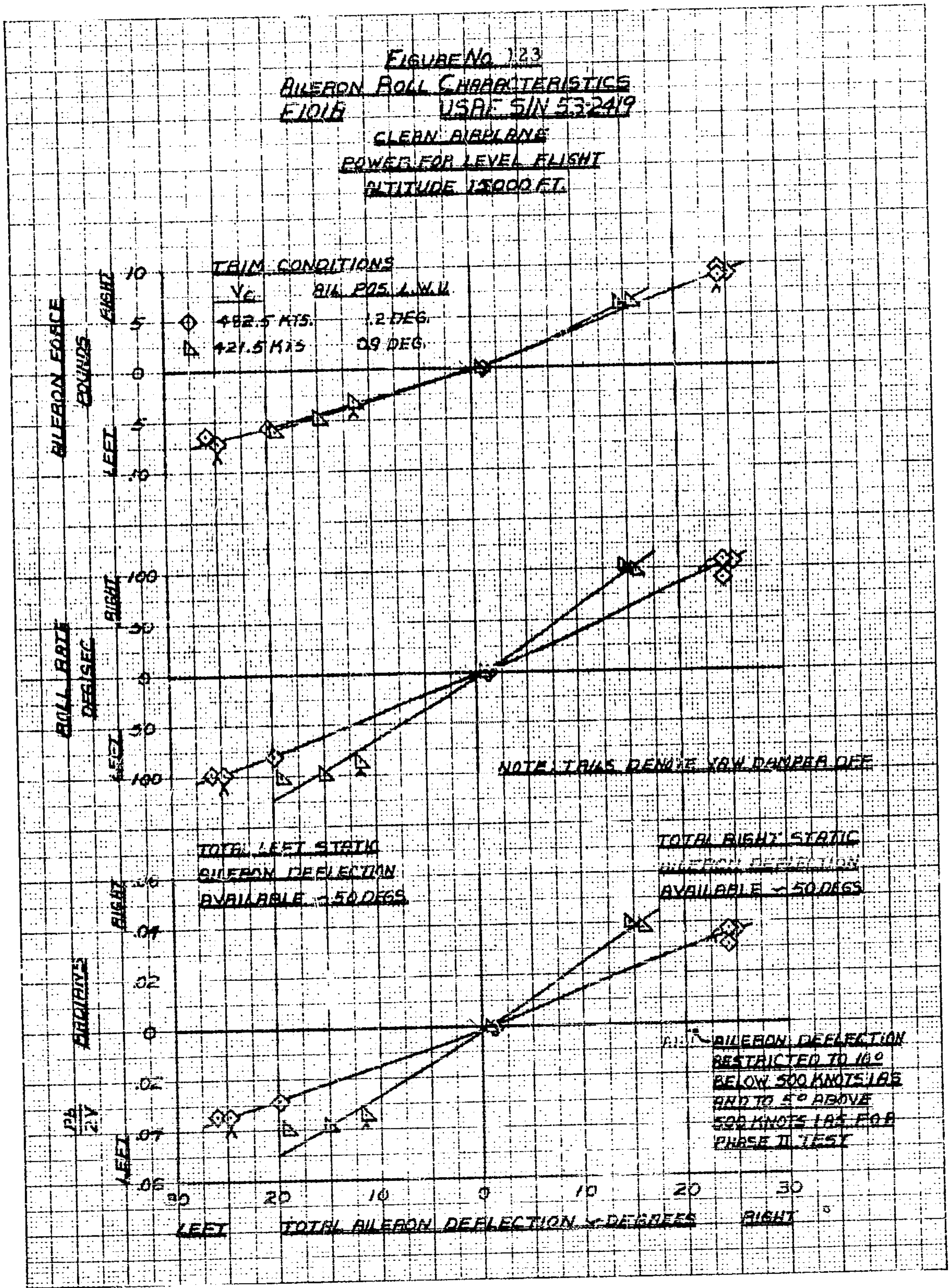


FIGURE No. 123
AILERON ROLL CHARACTERISTICS
F101B USAF S/N 53-2419
CLEAN AIRPLANE
POWER FOR LEVEL FLIGHT
ALTITUDE 15000 FT.
TRIM CONDITIONS
V_c AIL POS. L.W.U.
482.5 KTS. 1.2 DEG.
421.5 KTS 0.9 DEG.
AILERON FORCE POUNDS
RIGHT
LEFT
ROLL RATE DEG/SEC
NOTE: TAILS DENOTE YAW DAMPER OFF
TOTAL LEFT STATIC AILERON DEFLECTION AVAILABLE ~ 50 DEGS
TOTAL RIGHT STATIC AILERON DEFLECTION AVAILABLE ~ 50 DEGS
RADIANS
$\frac{pb}{2V}$
AILERON DEFLECTION RESTRICTED TO 10° BELOW 500 KNOTS IAS AND TO 5° ABOVE 500 KNOTS IAS FOR PHASE II TEST
LEFT TOTAL AILERON DEFLECTION ~ DEGREES RIGHT

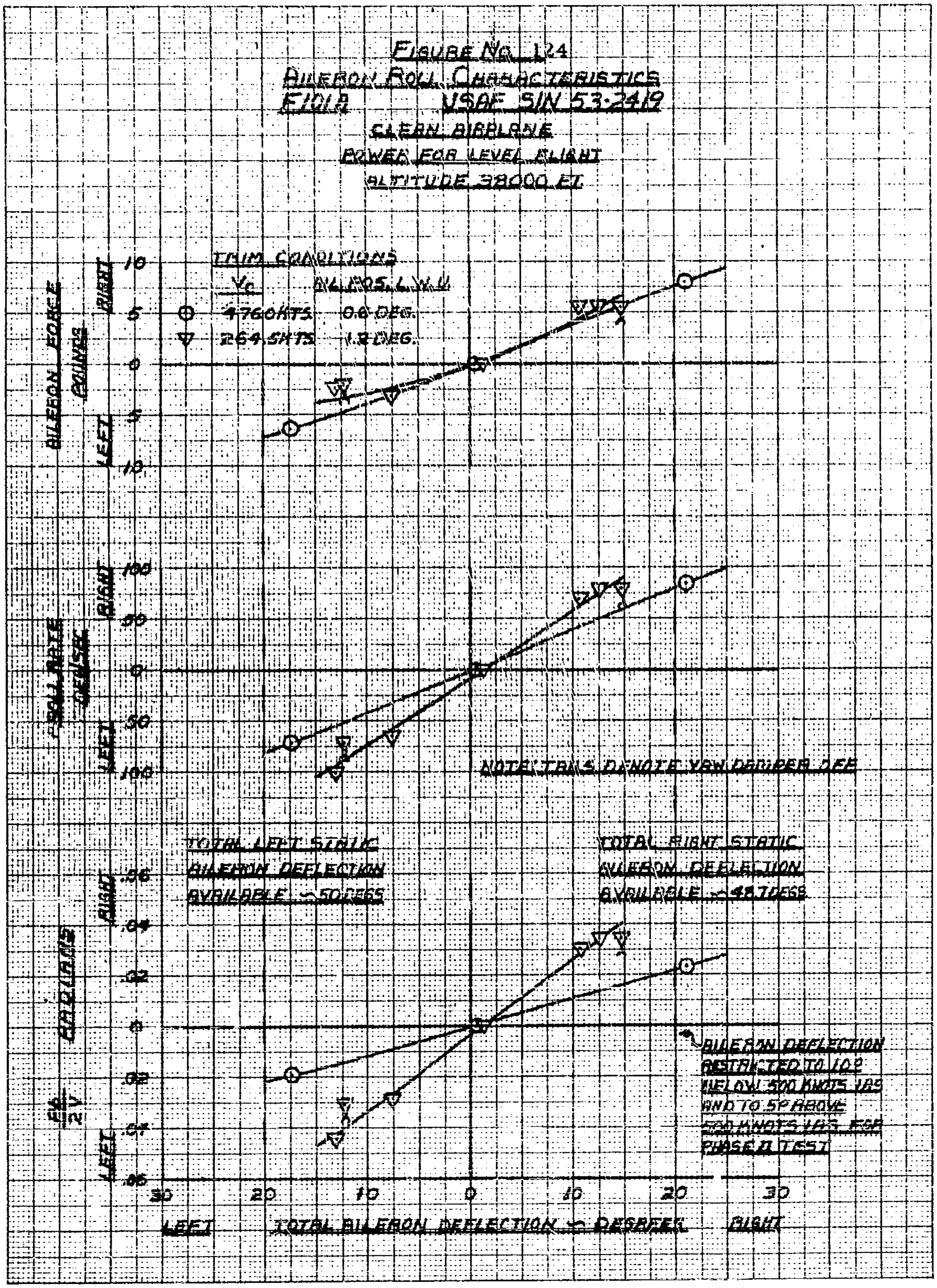


FIGURE NO. 124
AILERON ROLL CHARACTERISTICS
F101A USAF S/N 53-2419
CLEAN AIRPLANE
POWER FOR LEVEL FLIGHT
ALTITUDE 38000 FT.
TRIM CONDITIONS
Vc AIL POS. L.W.U.
476 KTS 0.6 DEG.
264.5 KTS 1.2 DEG.
AILERON FORCE POUNDS
RIGHT
LEFT
ROLL RATE DEG/SEC
NOTE: TAILS DENOTE YAW DAMPER OFF
TOTAL LEFT STATIC AILERON DEFLECTION AVAILABLE ~ 50 DEGS
TOTAL RIGHT STATIC AILERON DEFLECTION AVAILABLE ~ 45.7 DEGS
RADIANS
pb/2V
AILERON DEFLECTION RESTRICTED TO 10° BELOW 500 KNOTS IAS AND TO 5° ABOVE 500 KNOTS IAS FOR PHASE II TEST
LEFT
TOTAL AILERON DEFLECTION ~ DEGREES
RIGHT

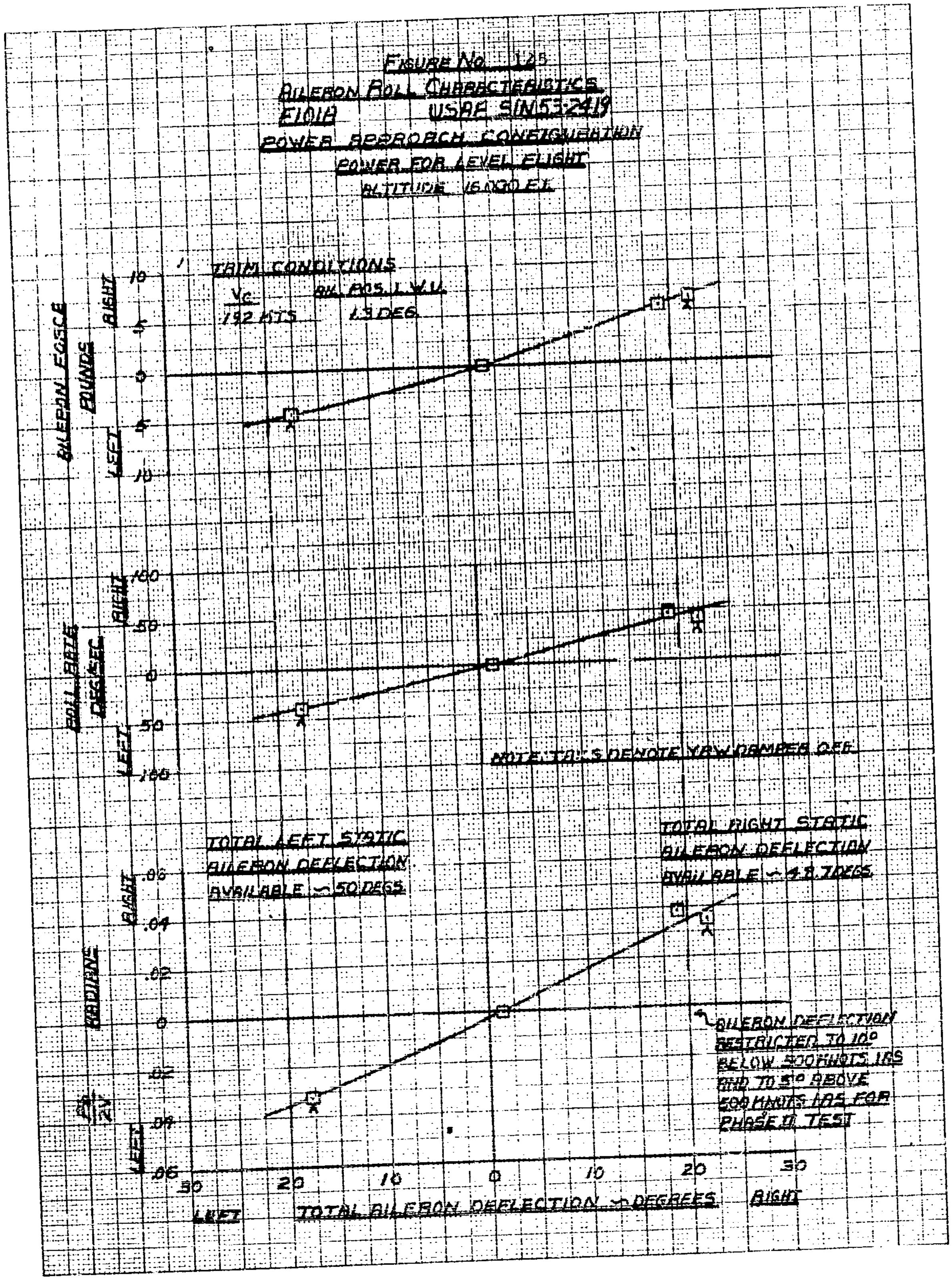


FIGURE NO. 125
AILERON ROLL CHARACTERISTICS
F101B USAF S/N53-2419
POWER APPROACH CONFIGURATION
POWER FOR LEVEL FLIGHT
ALTITUDE 16,000 FT.
TRIM CONDITIONS
V_C AIL POS. T.W.U.
192 KTS 1.3 DEG
AILERON FORCE POUNDS
RIGHT
LEFT
10
5
0
5
10
ROLL RATE DEG/SEC
100
50
0
50
100
NOTE: TAILS DENOTE YAW DAMPER OFF
TOTAL LEFT STATIC AILERON DEFLECTION AVAILABLE ~ 50 DEGS
TOTAL RIGHT STATIC AILERON DEFLECTION AVAILABLE ~ 48.7 DEGS
RADIANS
$\frac{pb}{2V}$
.06
.04
.02
0
.02
.04
.06
AILERON DEFLECTION RESTRICTED TO 10° BELOW 500 KNOTS IAS AND TO 5° ABOVE 500 KNOTS IAS FOR PHASE II TEST
30
20
10
0
10
20
30
LEFT
TOTAL AILERON DEFLECTION ~ DEGREES
RIGHT

Figure No. 126

MAXIMUM ALLOWABLE DEFLECTION*

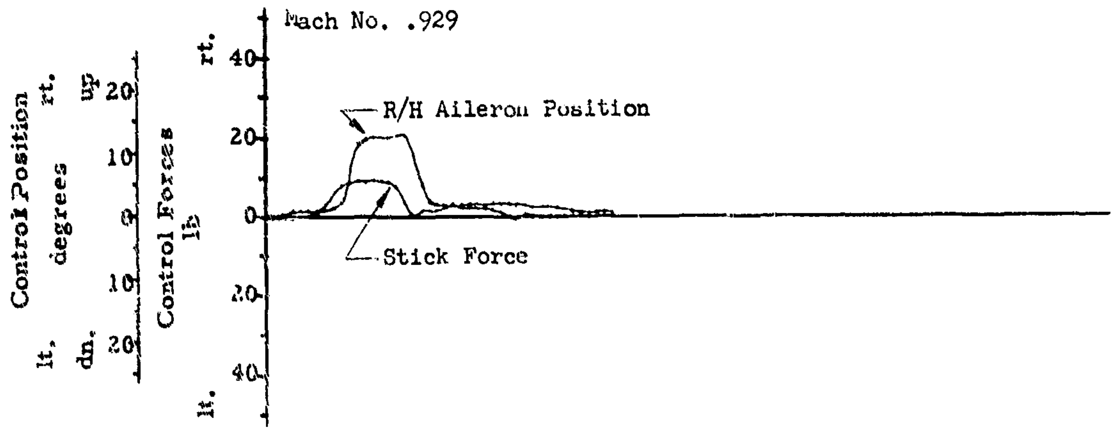
AILERON ROLLS

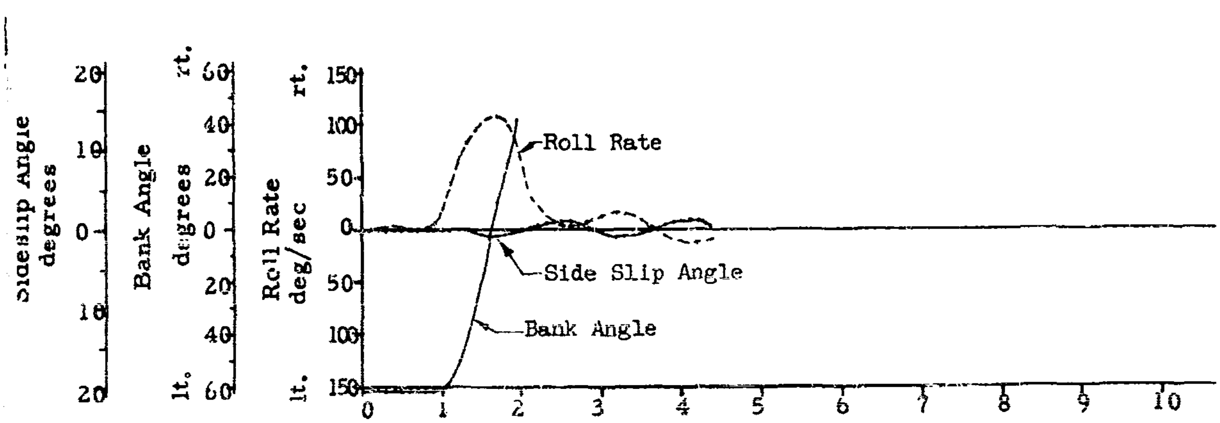
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	482.5	kts	CG pos.	31.4	%MAC
Alt.	14,710	ft	Weight	38,060	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	1.5° Nose Down		Rudder	Damper On	





APPENDIX I

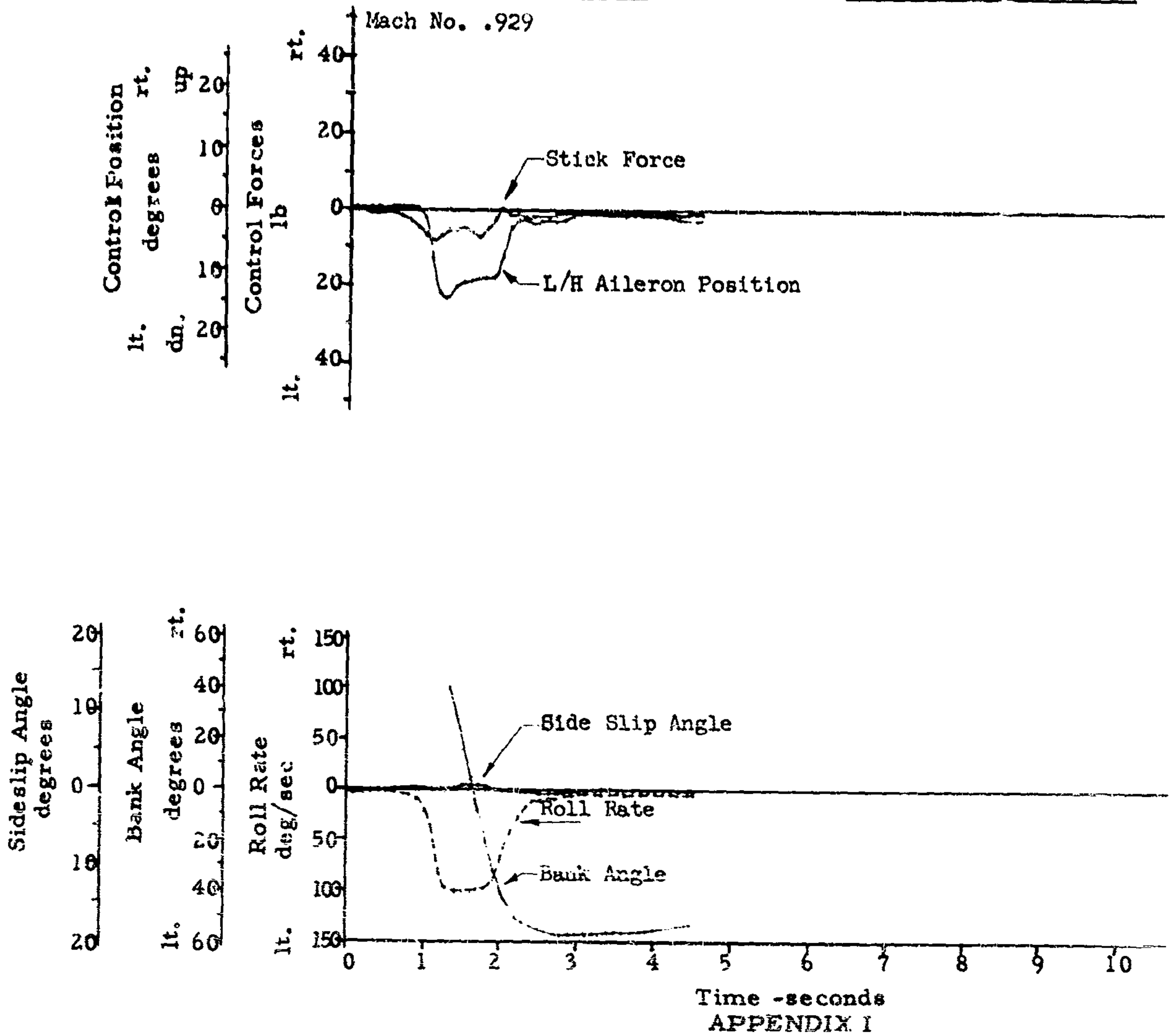
Figure No. 127

MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	482.5	kts	CG pos.	31.4	%MA
Alt.	14,710	ft	Weight	38,060	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	1.5° Nose Down		Rudder	Damper On	



APPENDIX I

Figure No. 128

MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

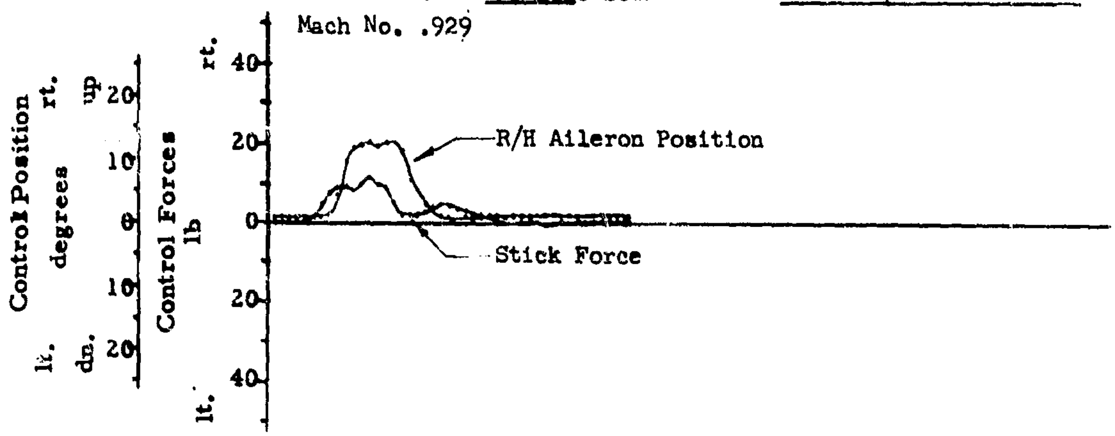
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	482.5	kts	CG pos.	31.4	%MAC
Alt.	14,710	ft	Weight	38,060	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	1.5° Nose Down		Rudder	Damper Off	

Mach No. .929



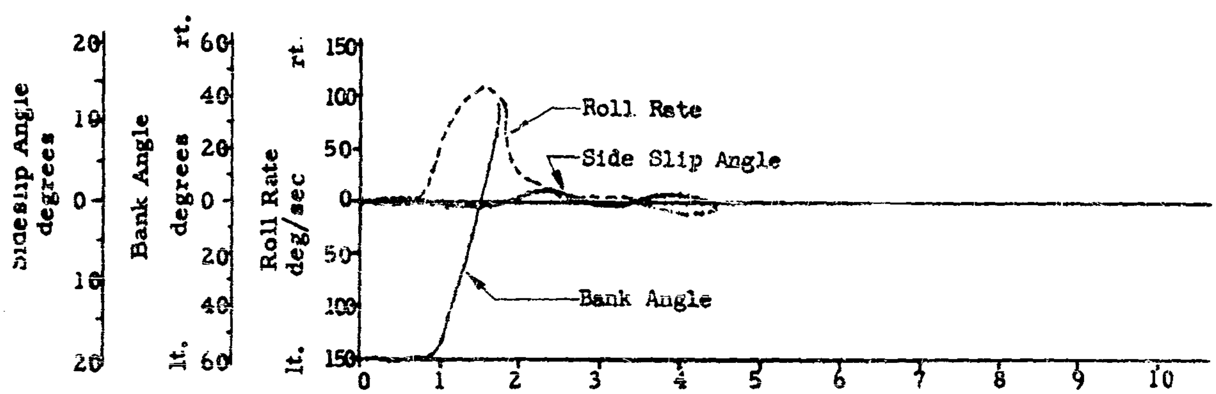


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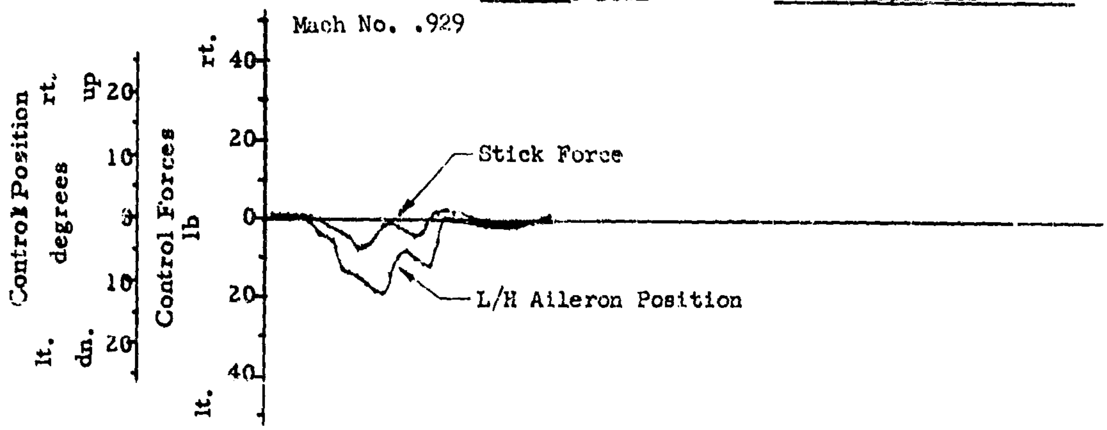
Figure No. 129

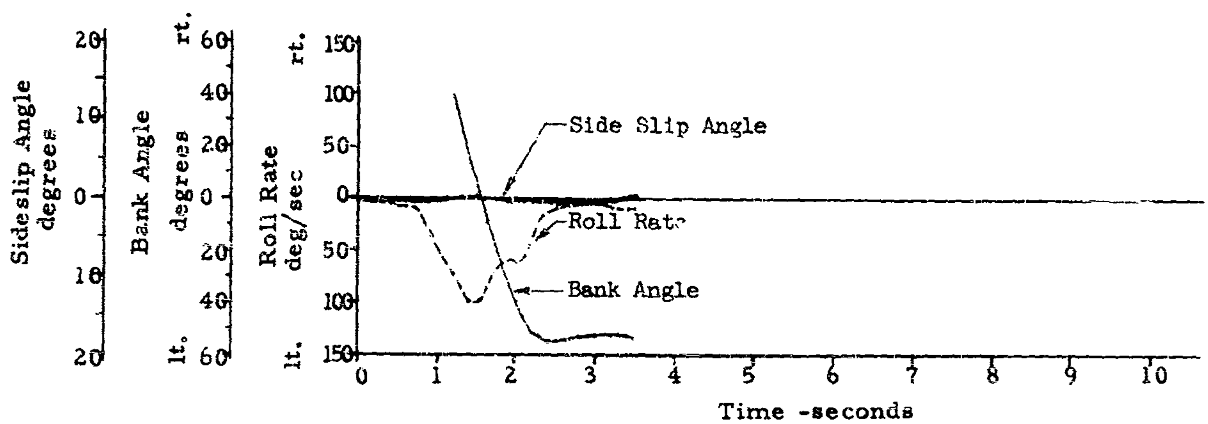
MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 482.5 kts CG pos. 31.4 %MAC
Alt. 14,710 ft Weight 38,060 lb
Rudder pos. 0 Deg. Aileron pos. 1.2 Deg. L.W.U.
Stabilizer pos. 1.5° Nose Down Rudder Damper Off





APPENDIX I

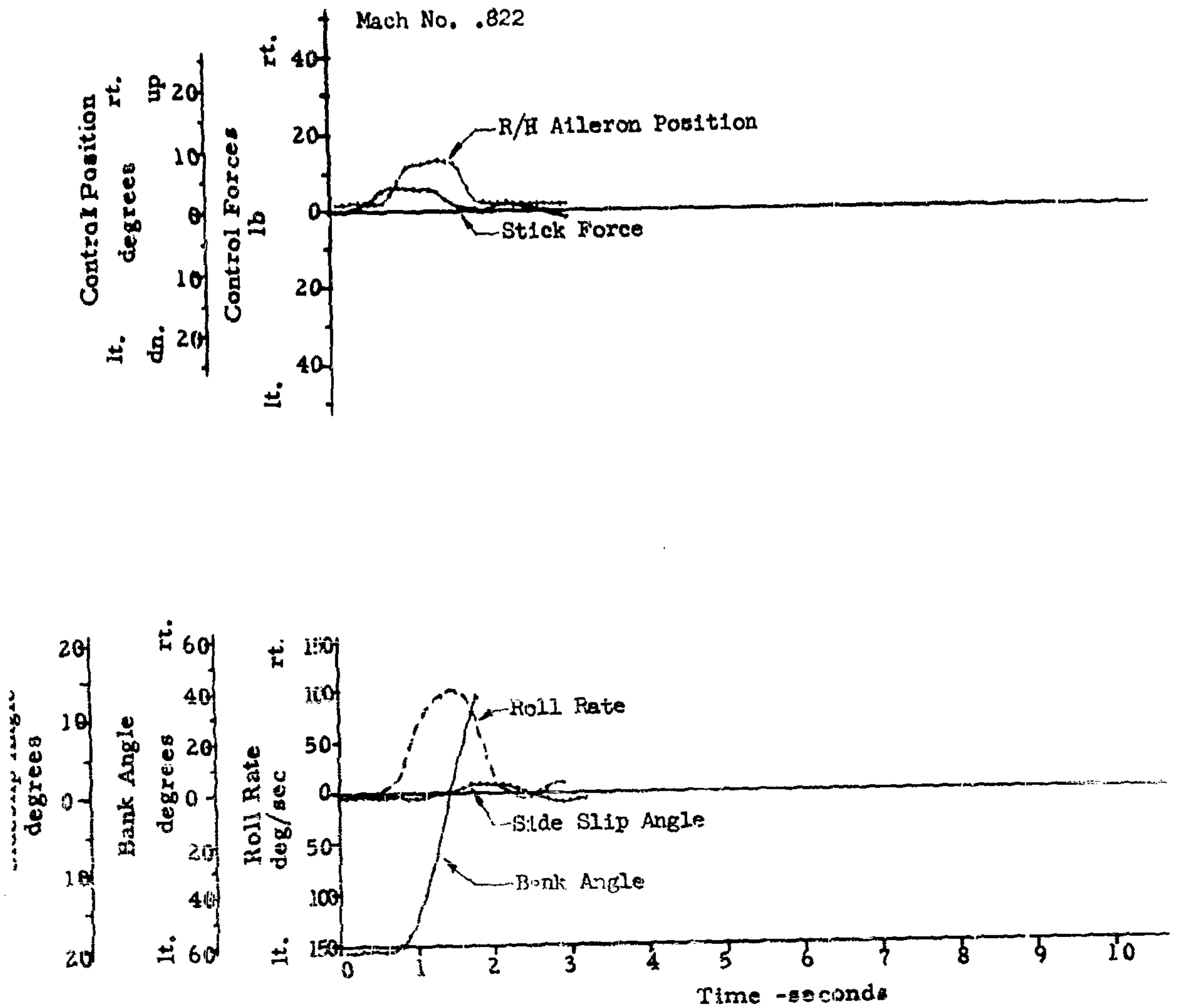
Figure No. 130

MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	421.5	kts	CG pos.	31.3	%MAC
Alt.	15,040	ft	Weight	36,800	lb
Rudder pos.	0 Deg.		Aileron pos.	.9 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper On	



APPENDIX I

Figure No. 132

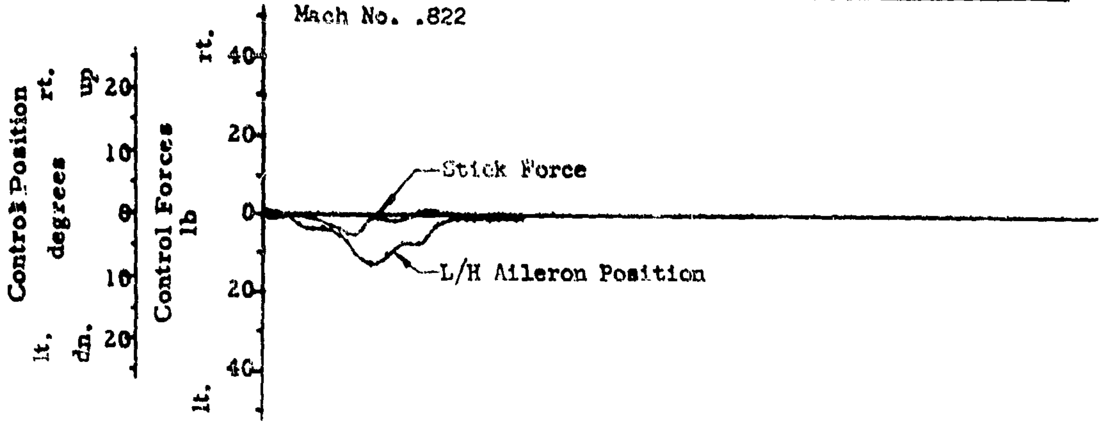
MAXIMUM ALLOWABLE DEFLECTION*

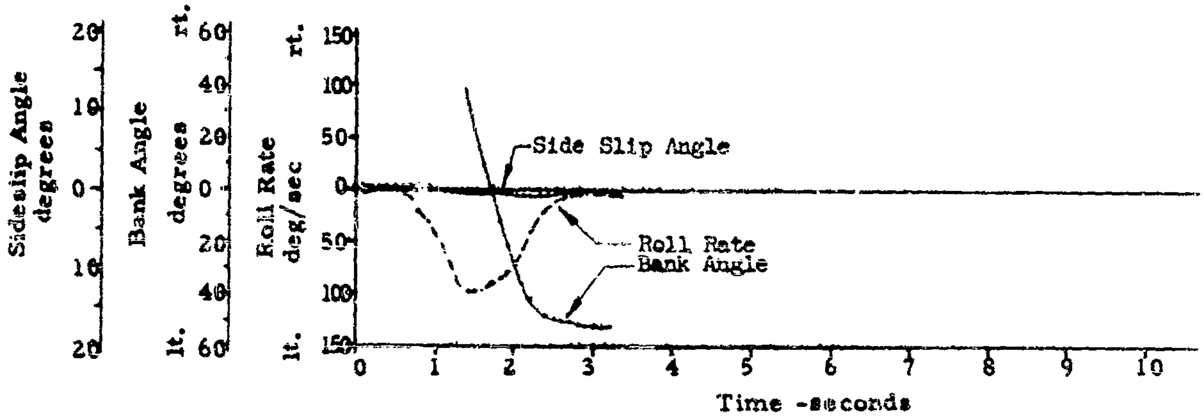
AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	421.5	kts	CG pos.	31.3	%MAC
Alt.	15,040	ft	Weight	36,800	lb
Rudder pos.	0 Deg.		Aileron pos.	0.9 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper Off	

Mach No. .822





APPENDIX I

Figure No. 131

MAXIMUM ALLOWABLE DEFLECTION*

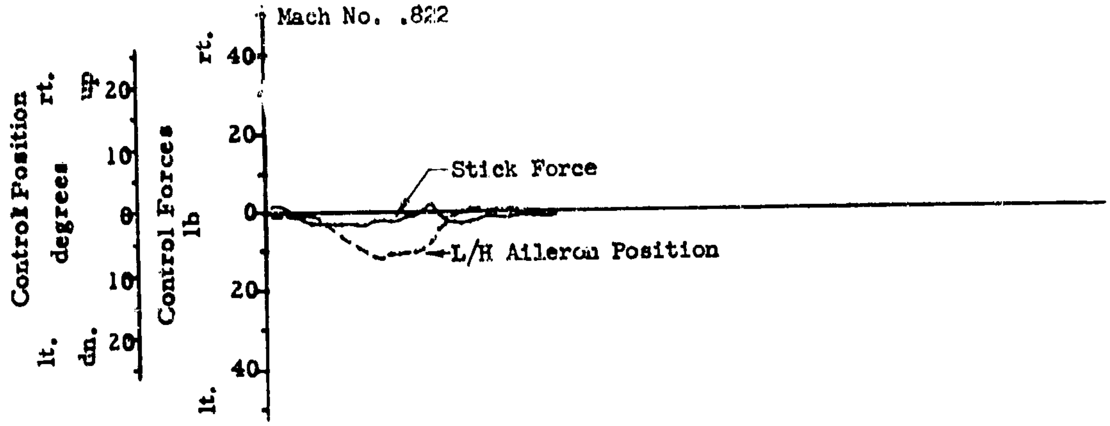
AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration

TRIM CONDITIONS

CAS	421.5	kts	CG pos.	31.3	%MAC
Alt.	15,040	ft	Weight	36,800	lb
Rudder pos.	0 Deg.		Aileron pos.	0.9 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper On	

Mach No. .822



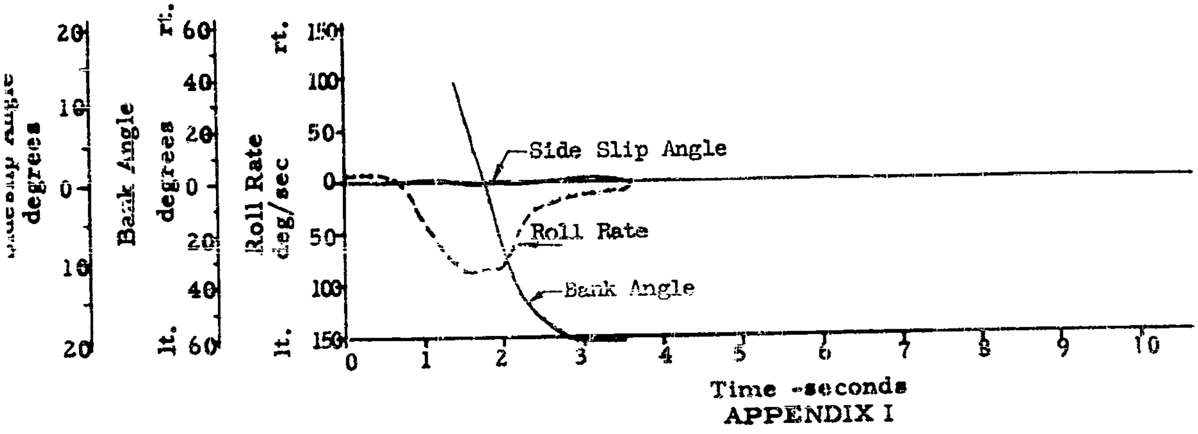


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Figure No. 133

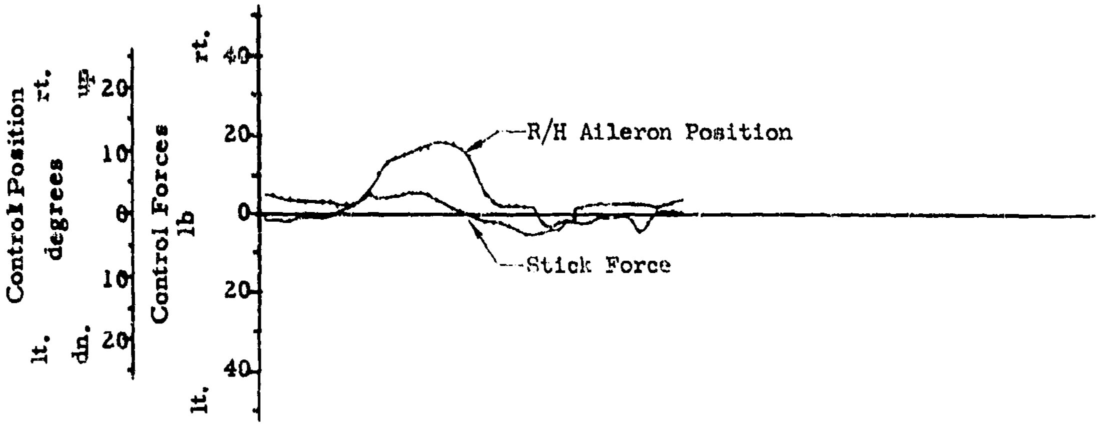
MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Power Approach Configuration

TRIM CONDITIONS

CAS	192	kts	CG pos.	32.2	%MAC
Alt.	16,020	ft	Weight	33,850	lb
Rudder pos.	0 Deg.		Aileron pos.	1.5 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper On	



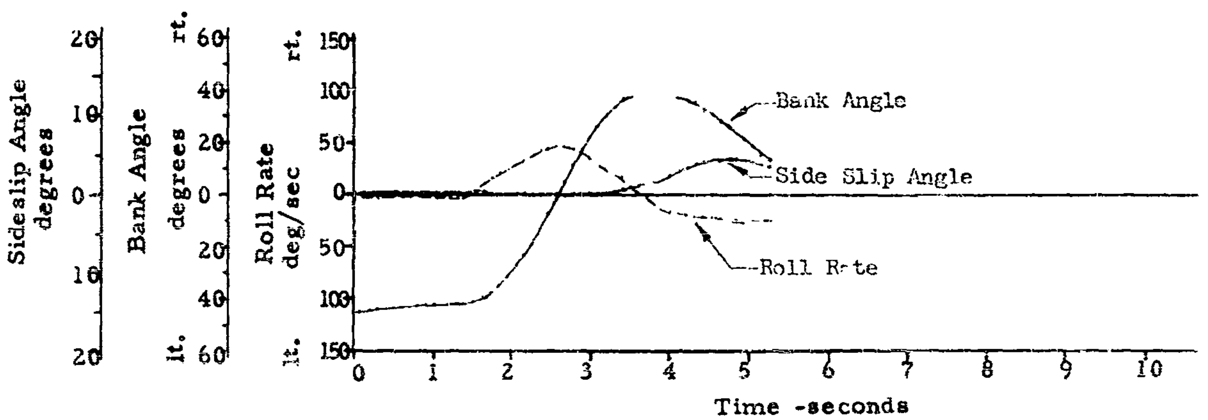


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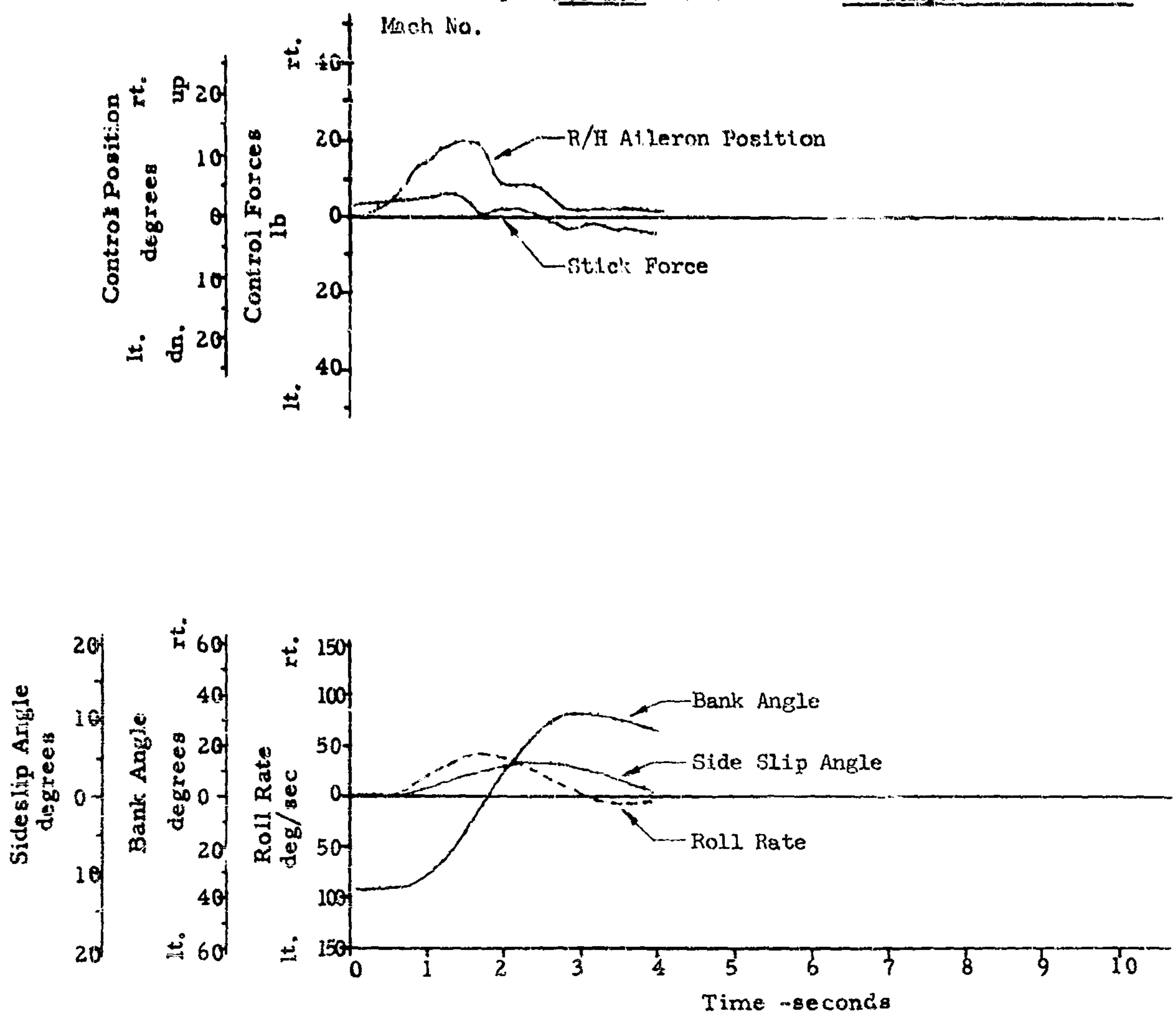
Figure No. 134

MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Power Approach Configuration
TRIM CONDITIONS

CAS	192	kts	CG pos.	32.2	%MAC
Alt.	16,020	ft	Weight	33,850	lb
Rudder pos.	0 Deg.		Aileron pos.	1.3 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper Off	



APPENDIX I

Figure No. 136

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

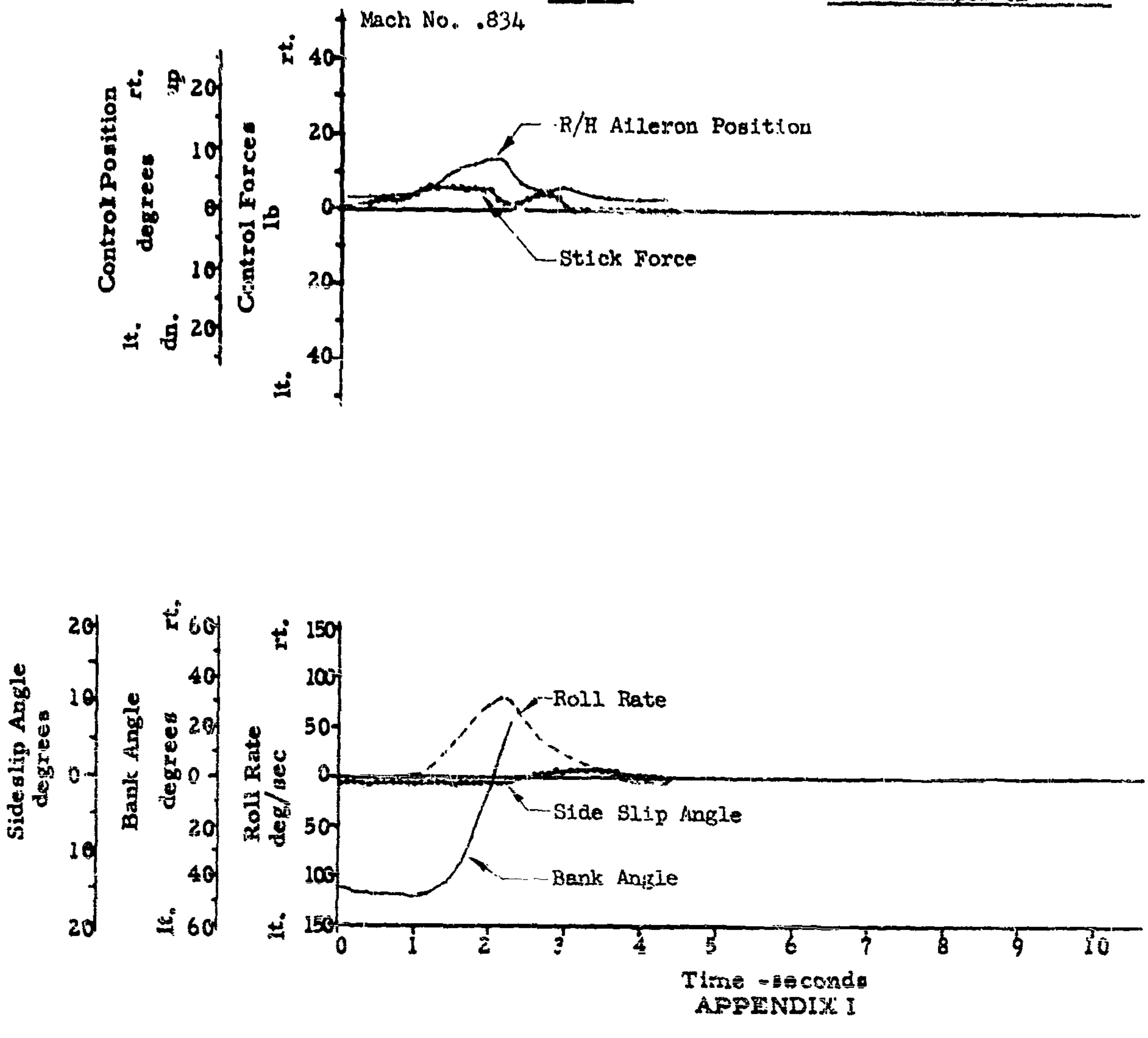
CAS 264.5 kts — CG pos. 34.3 %MAC

Alt. 38,150 ft — Weight 31,640 lb

Rudder pos. 0 Deg. — Aileron pos. 1.2 Deg. L.W.U.

Stabilizer pos. 0 Deg. — Rudder Damper On

Mach No. .834



APPENDIX I

Figure No. 135

MAXIMUM ALLOWABLE DEFLECTION

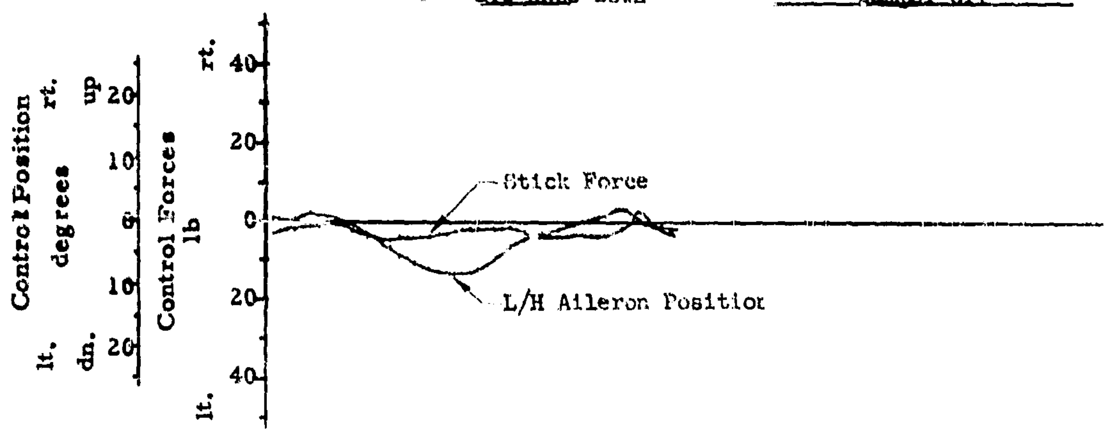
AILERON ROLLS

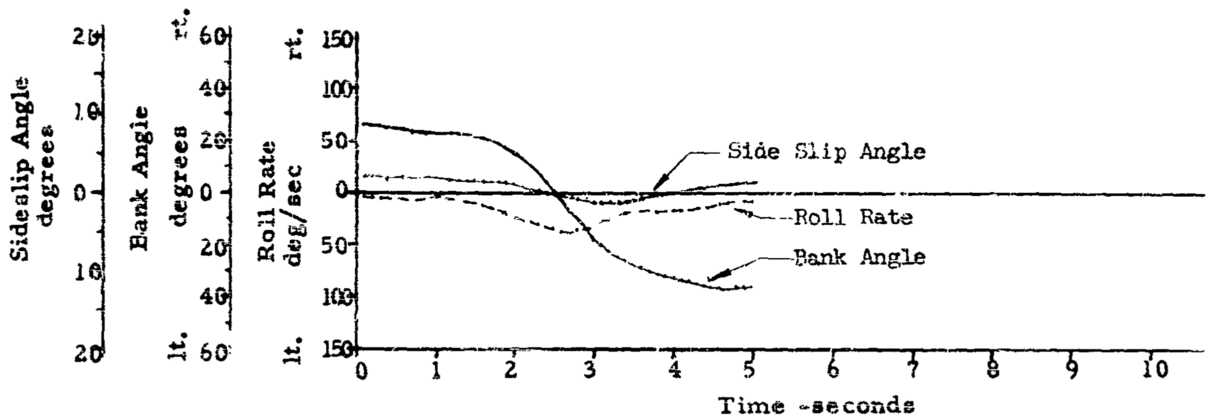
F-101A, USAF No. 53-2419

Power Approach Configuration

TRIM CONDITIONS

CAS	192	kts	CG pos.	32.2	%MAC
Alt.	16,020	ft	Weight	33,850	lb
Rudder pos.	0 Deg.		Aileron pos.	1.3 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper Off	





APPENDIX I

Figure No. 137

MAXIMUM ALLOWABLE DEFLECTION*

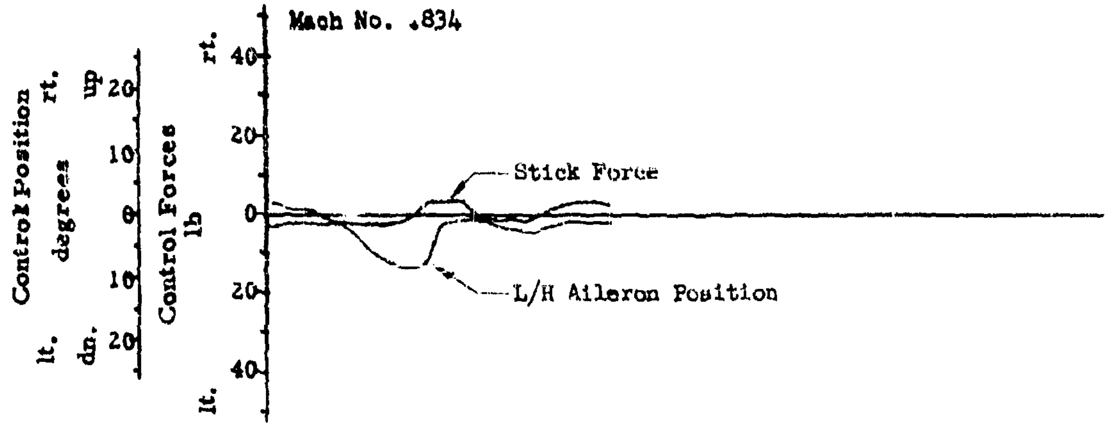
AILERON ROLLS

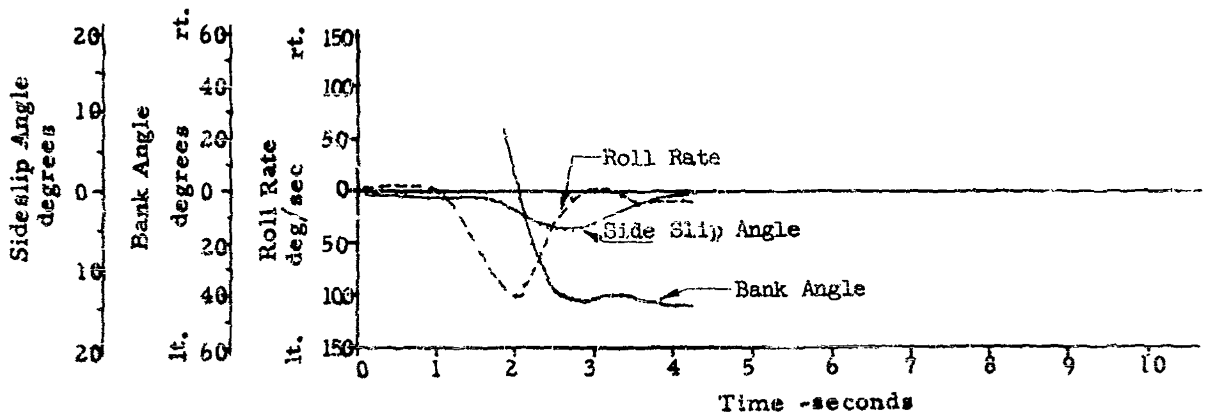
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS 264.5 kts   CG pos. 34.3 %MAC
Alt. 38,150 ft   Weight 31,640 lb
Rudder pos. 0 Deg.   Aileron pos. 1.2 Deg. L.W.U.
Stabilizer pos. 0 Deg.   Rudder Damper On





APPENDIX I

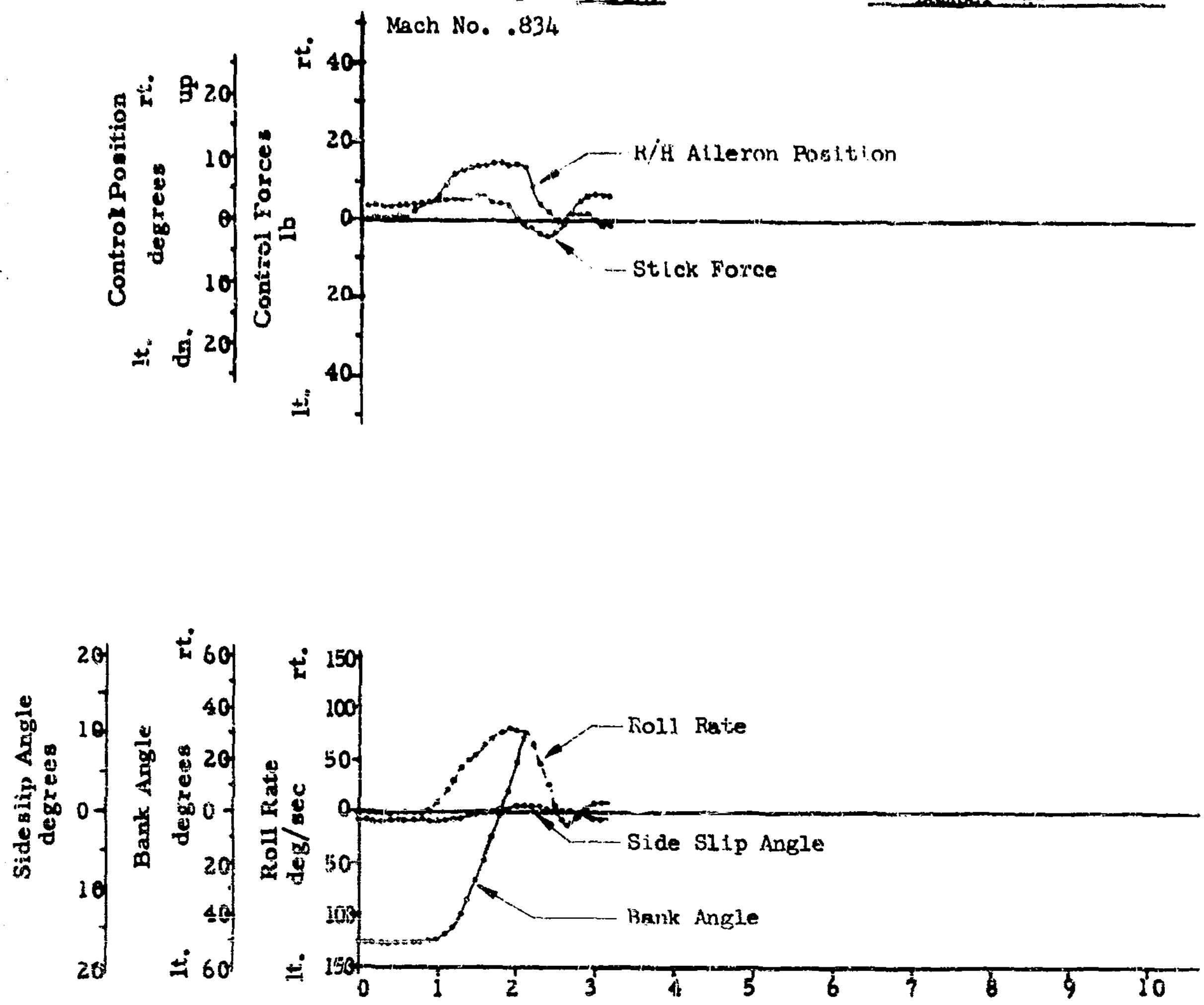
Figure No. 138

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 264.5 kts CG pos. 34.3 %MAC
Alt. 38,150 ft Weight 31,640 lb
Rudder pos. 0 Deg. Aileron pos. 1.2 Deg. L.W.U.
Stabilizer pos. 0 Deg. Rudder Damper Off



APPENDIX I

Figure No. 139

MAXIMUM ALLOWABLE DEFLECTION

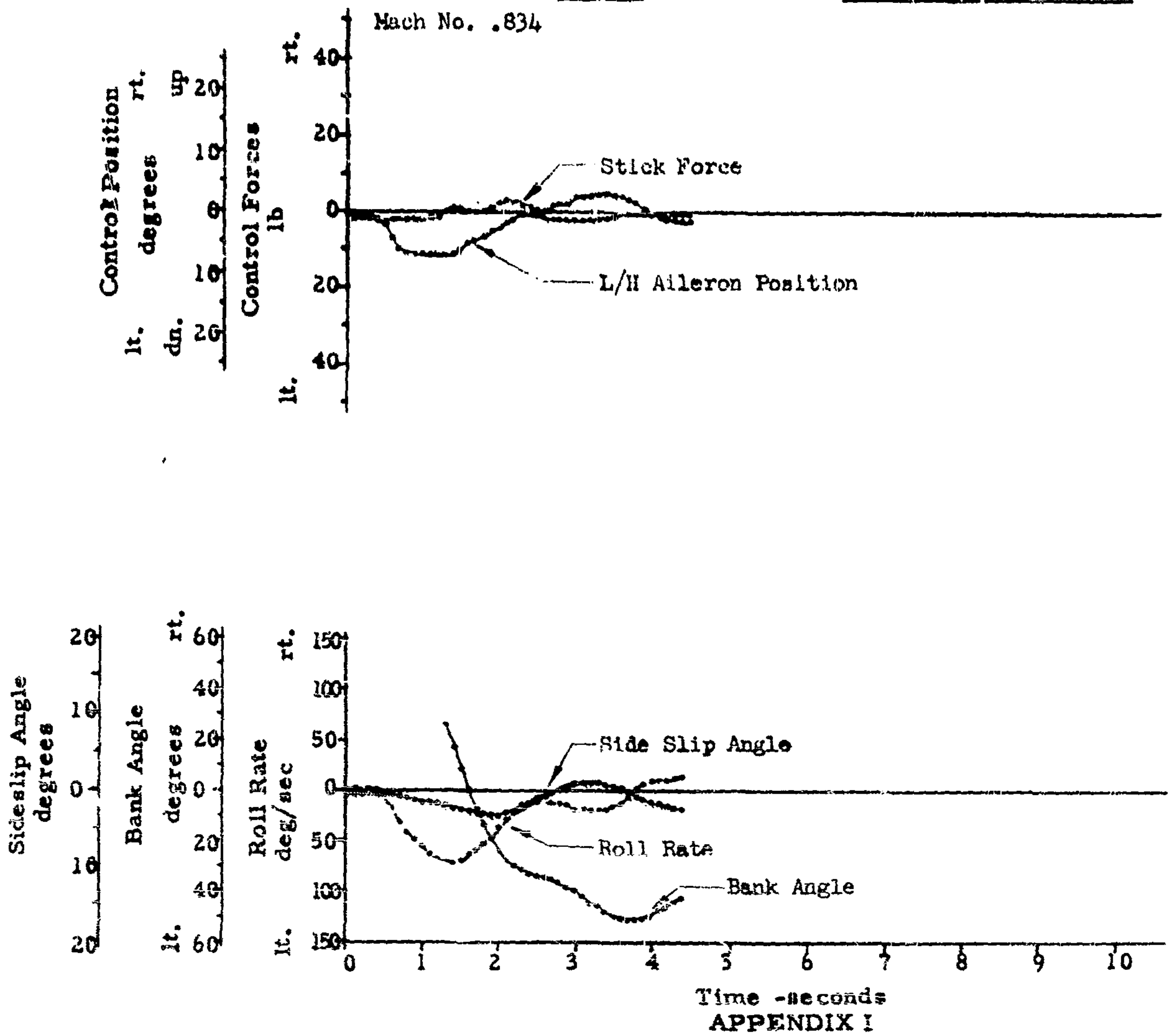
AILERON ROLLS

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS 264.5 kts   CG pos. 34.3 %MAC
Alt. 38,150 ft   Weight 31,640 lb
Rudder pos. 0 Deg.   Aileron pos. 1.0 Deg. L.W.U.
Stabilizer pos. 0 Deg.   Rudder Damper Off



APPENDIX I

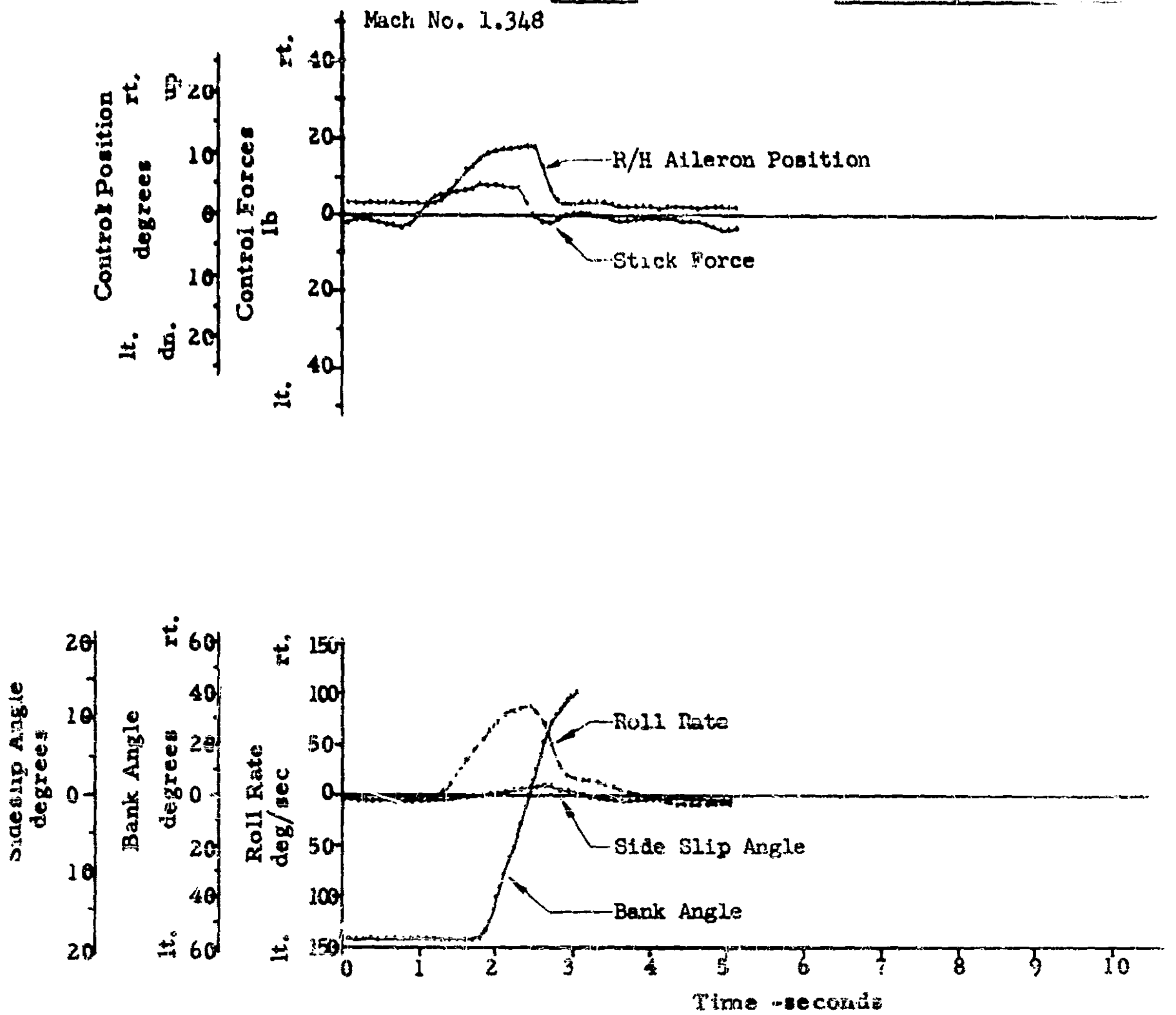
Figure No. 140

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

F-101A, USAF No. 53-2419
Combat Configuration
TRIM CONDITIONS

CAS	476	kts	CG pos.	33.6	%MAC
Alt.	36,360	ft	Weight	34,470	lb
Rudder pos.	0 Deg.		Aileron pos.	.6 Deg. L.W.U.	
Stabilizer pos.	0 Deg.		Rudder	Damper On	



APPENDIX I

Figure No. 141

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

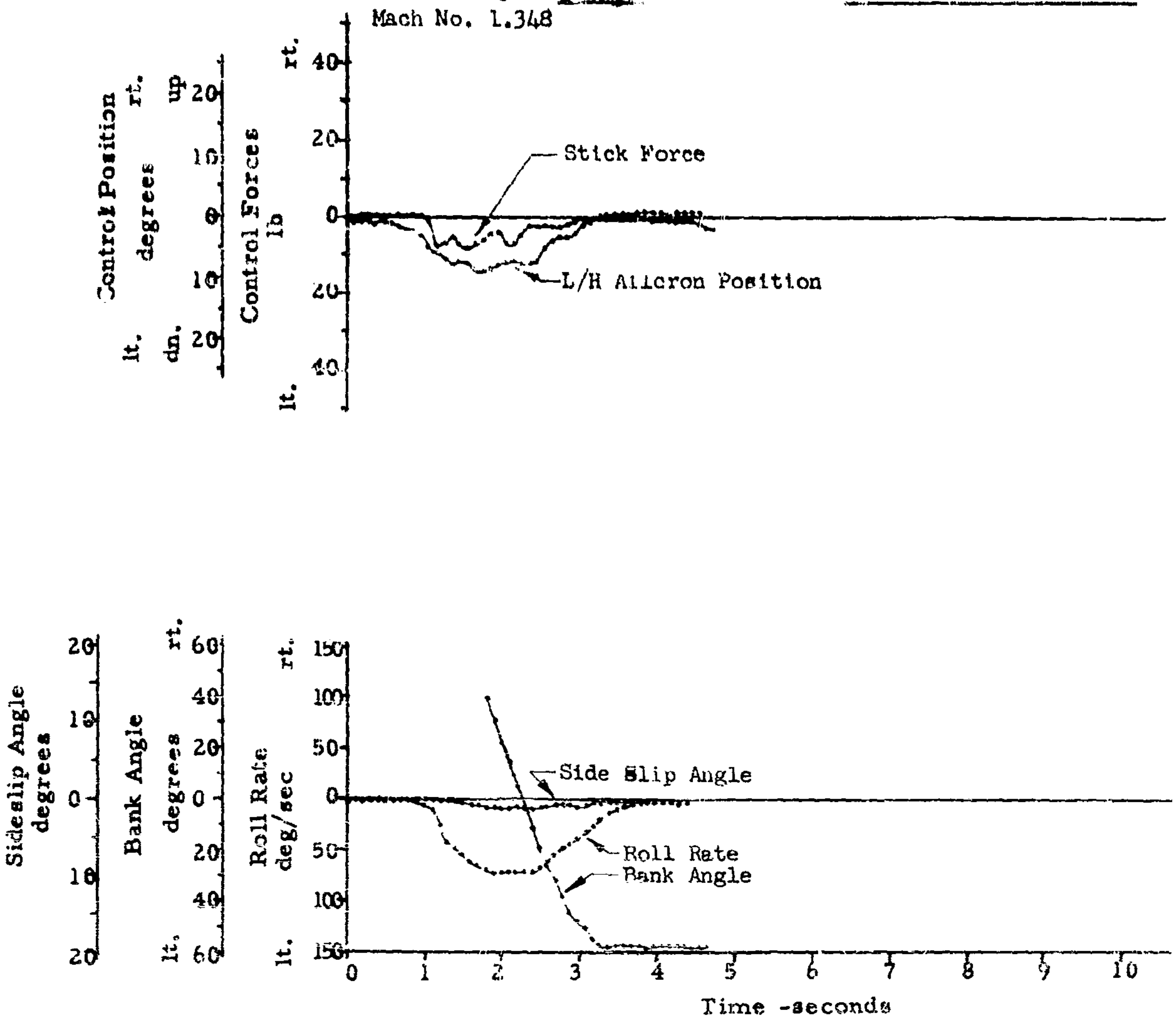
F-101A, USAF No. 53-2419

Combat Configuration

TRIM CONDITIONS

CAS	476	kts	CG pos.	33.6	%MAC
Alt.	36,360	ft	Weight	34,470	lb
Rudder pos.	0 Deg.		Aileron pos.	.6 Deg. L.W.U.	
Stabilizer pos.	0 Deg.		Rudder		

Mach No. 1.348



APPENDIX I

Figure No. 126

MAXIMUM ALLOWABLE DEFLECTION*

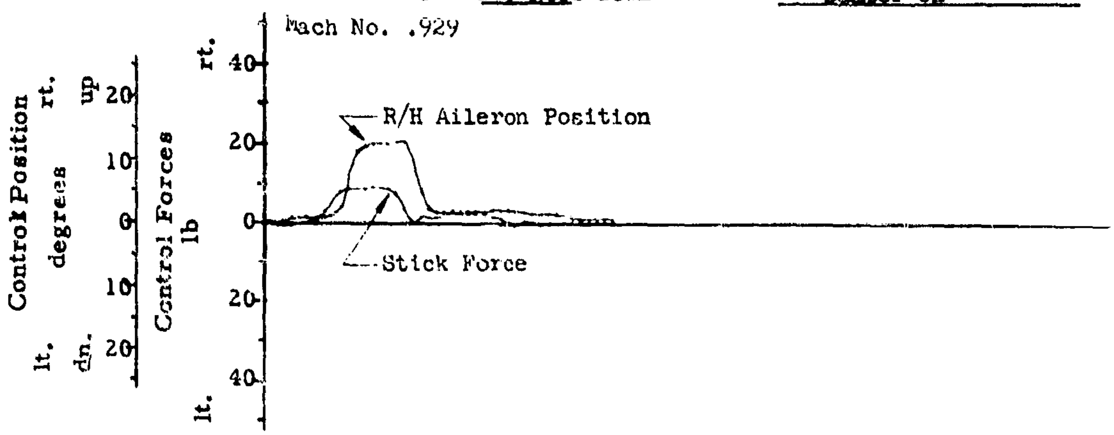
AILERON ROLLS

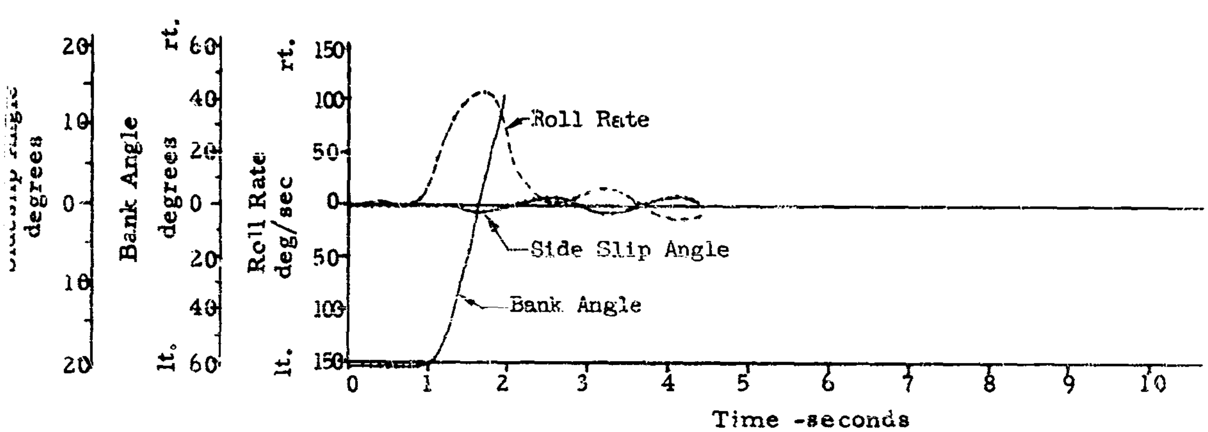
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	482.5	kts	CG pos.	31.4	%MAC
Alt.	14,710	ft	Weight	38,060	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	1.5° Nose Down		Rudder	Damper On	





APPENDIX 1

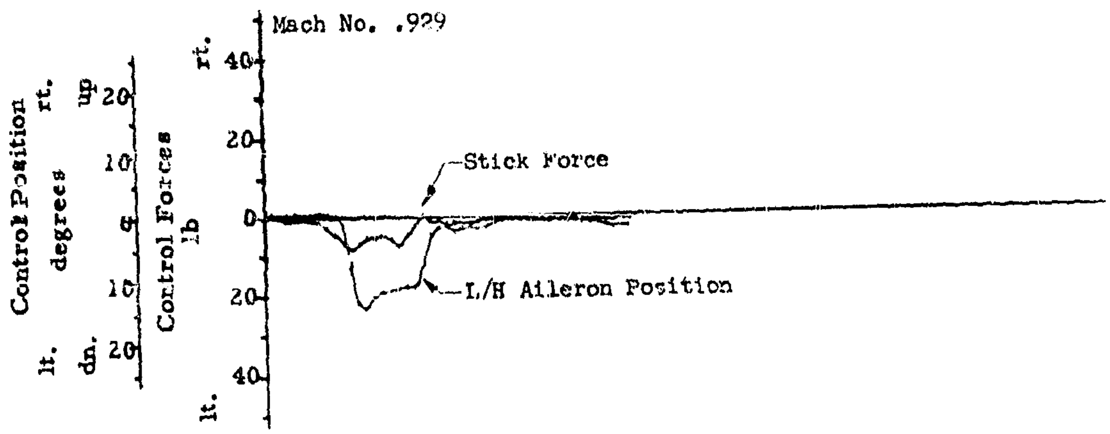
Figure No. 127

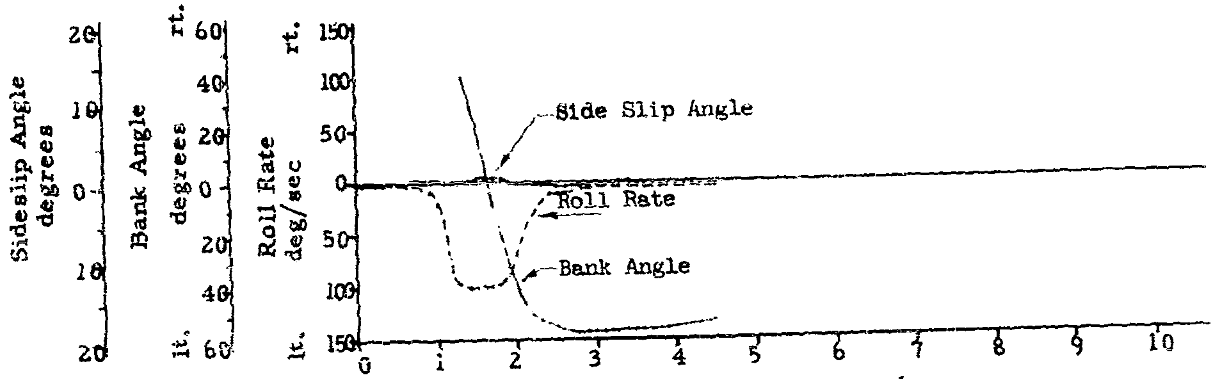
MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	482.5	kts	CG pos.	31.4	%MA
Alt.	14,710	ft	Weight	38,060	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	1.5° Nose Down		Rudder	Damper On	





APPENDIX I

Figure No. 128

MAXIMUM ALLOWABLE DEFLECTION*

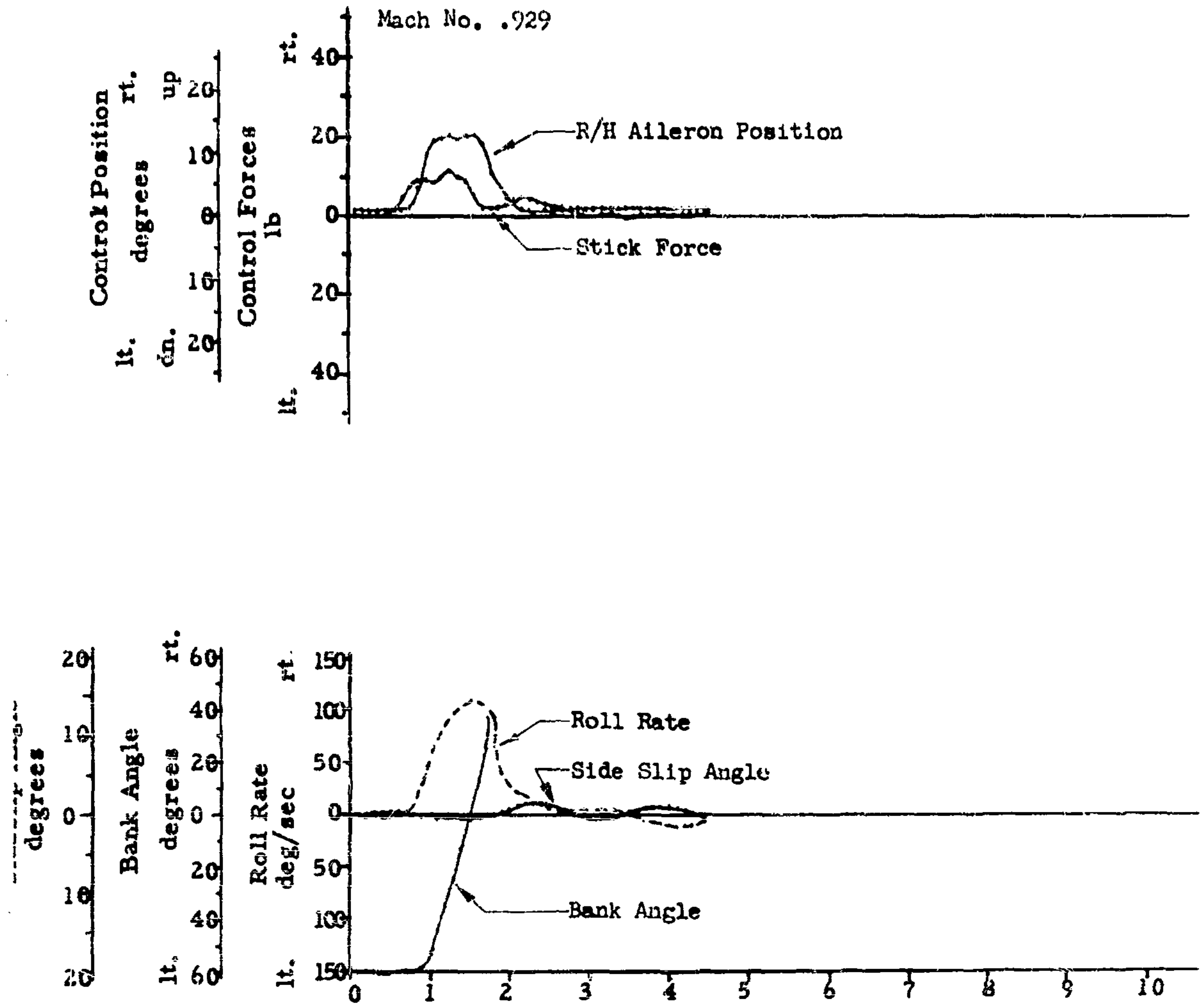
AILERON ROLLS

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	482.5	kts	CG pos.	31.4	%MAC
Alt.	14,710	ft	Weight	38,060	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	1.5° Nose Down		Rudder	Damper Off	



Time -seconds

APPENDIX I

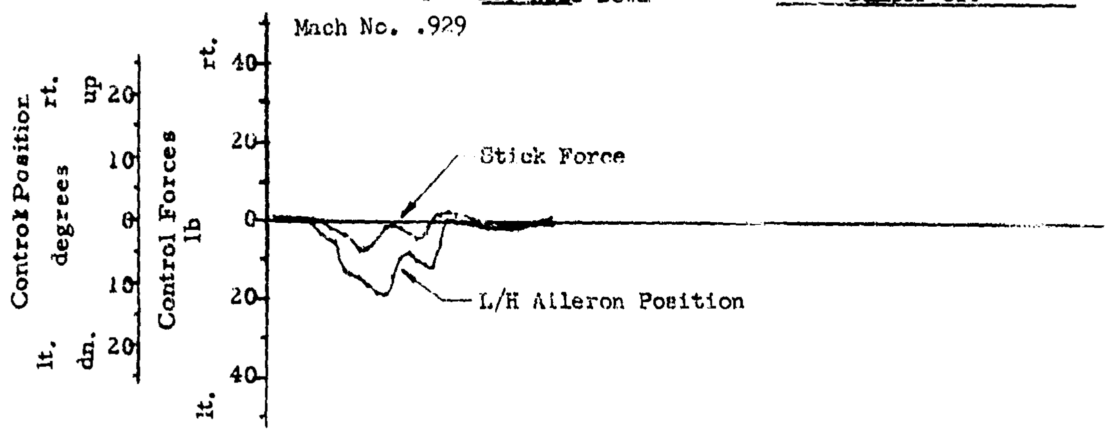
Figure No. 129

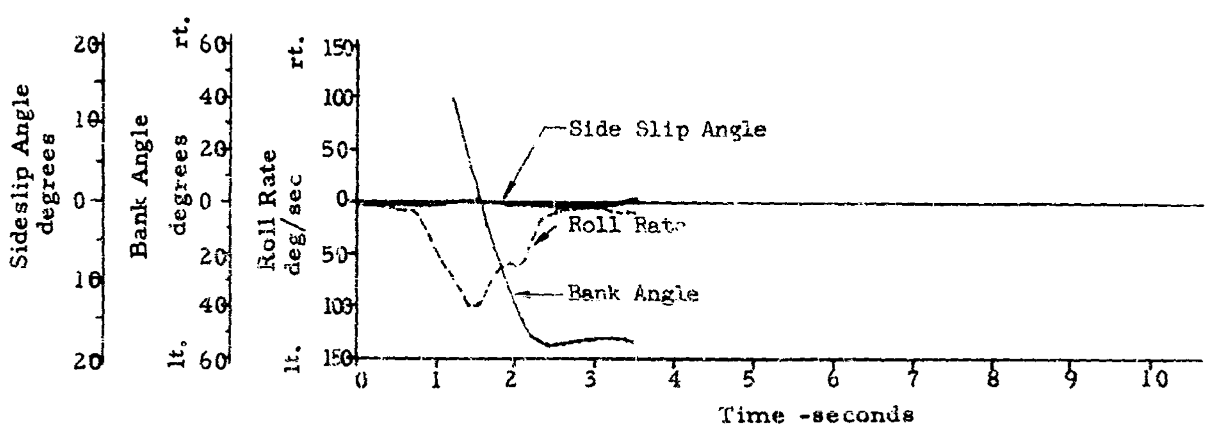
MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 482.5 kts CG pos. 31.4 %MAC
Alt. 14,710 ft Weight 38,060 lb
Rudder pos. 0 Deg. Aileron pos. 1.2 Deg. L.W.U.
Stabilizer pos. 1.5° Nose Down Rudder Damper Off





APPENDIX I

Figure No. 130

MAXIMUM ALLOWABLE DEFLECTION*

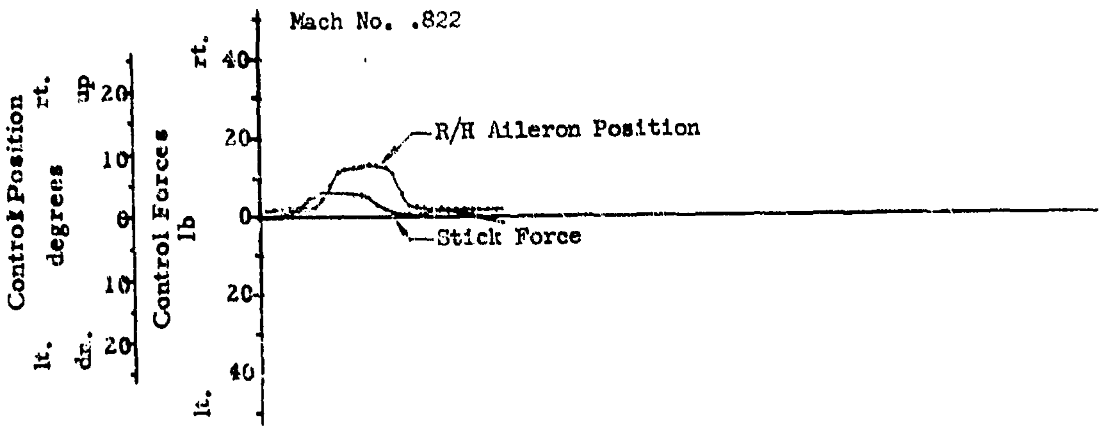
AILERON ROLLS

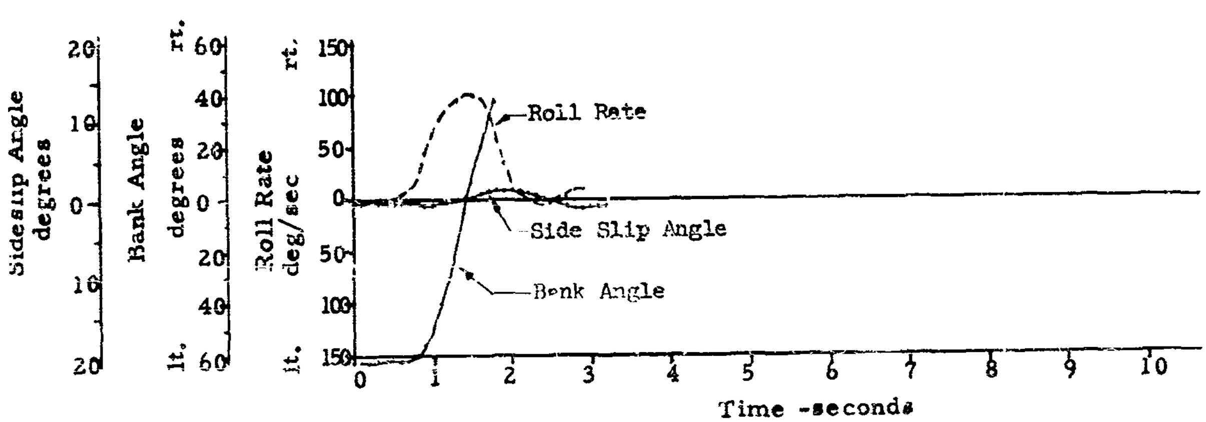
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	421.5	kts	CG pos.	31.3	%MAC
Alt.	15,040	ft	Weight	36,800	lb
Rudder pos.	0 Deg.		Aileron pos.	.9 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper On	





APPENDIX I

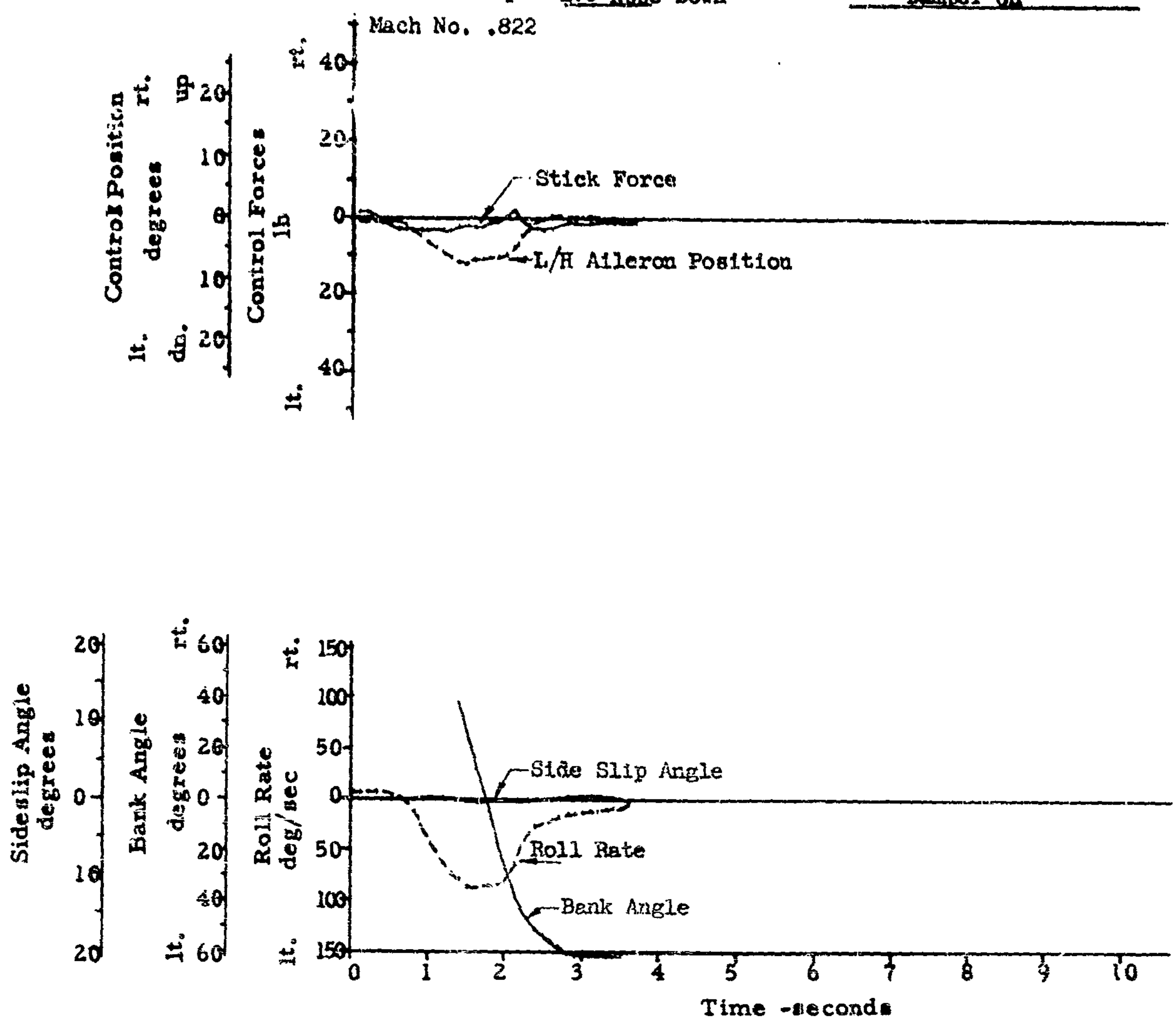
Figure No. 131

MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	421.5	kts	CG pos.	31.3	%MAC
Alt.	15,040	ft	Weight	36,800	lb
Rudder pos.	0 Deg.		Aileron pos.	0.9 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper On	



APPENDIX I

Figure No. 132

MAXIMUM ALLOWABLE DEFLECTION*

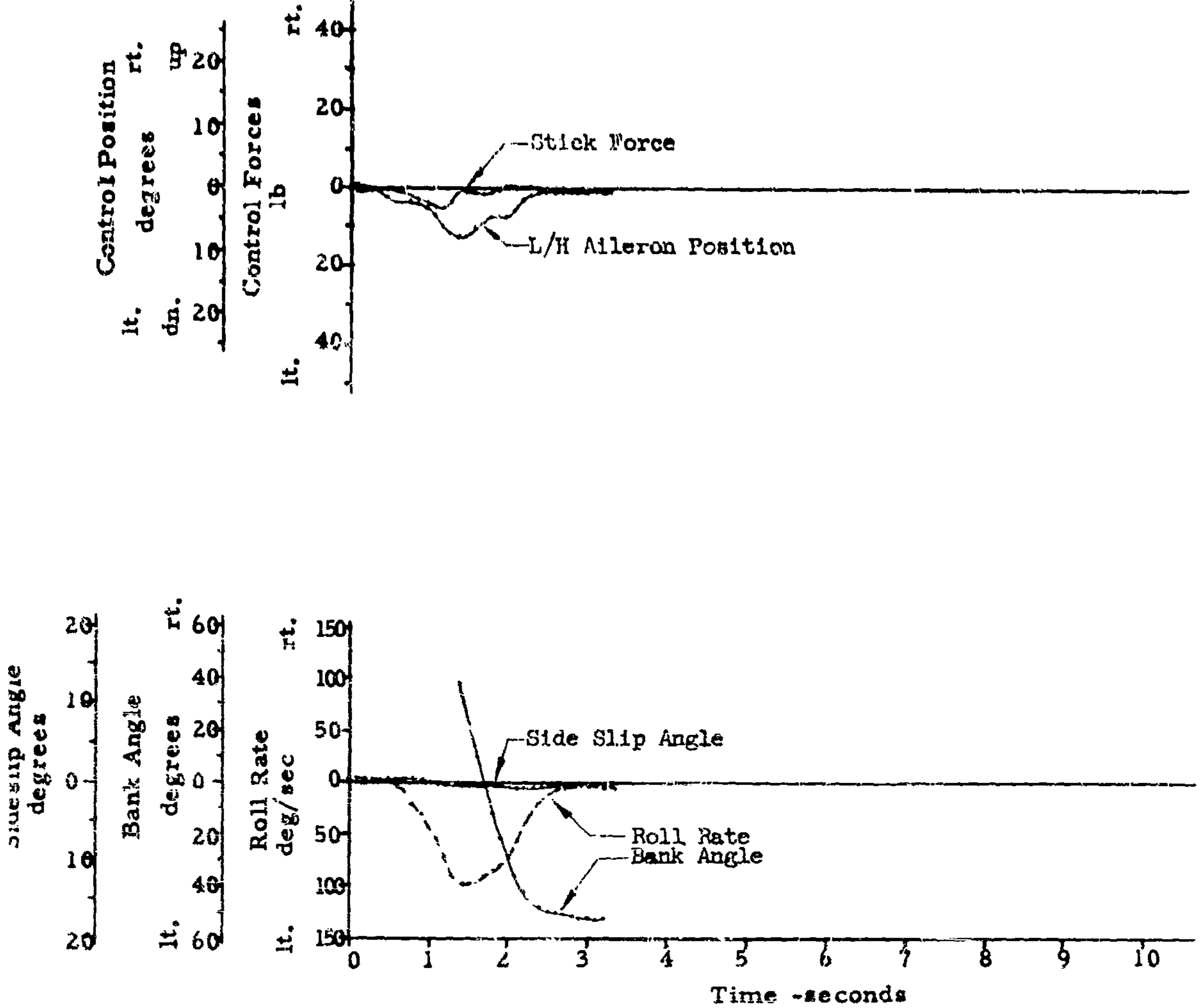
AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration

TRIM CONDITIONS

CAS	421.5	kts	CG pos.	31.3	%MAC
Alt.	15,040	ft	Weight	36,800	lb
Rudder pos.	0 Deg.		Aileron pos.	0.9 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper Off	

Mach No. .822



APPENDIX I

Figure No. 133

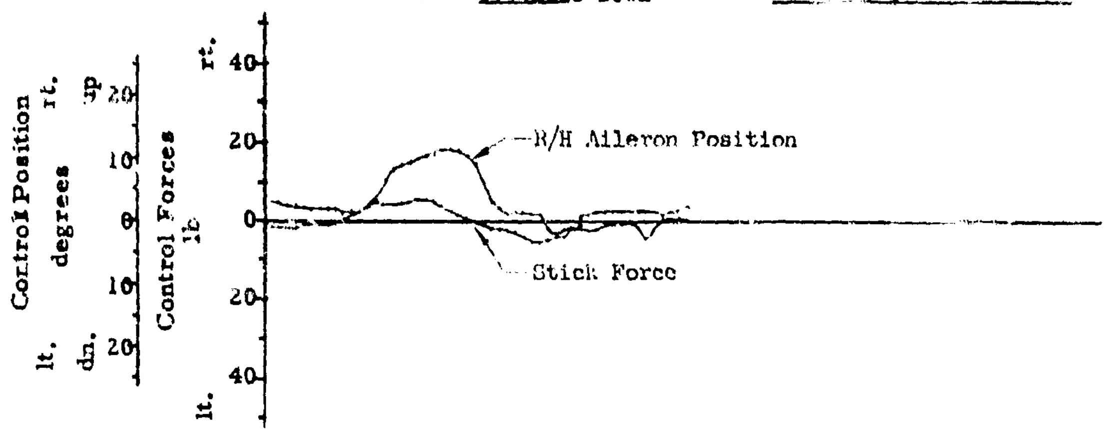
MAXIMUM ALLOWABLE DEFLECTION*

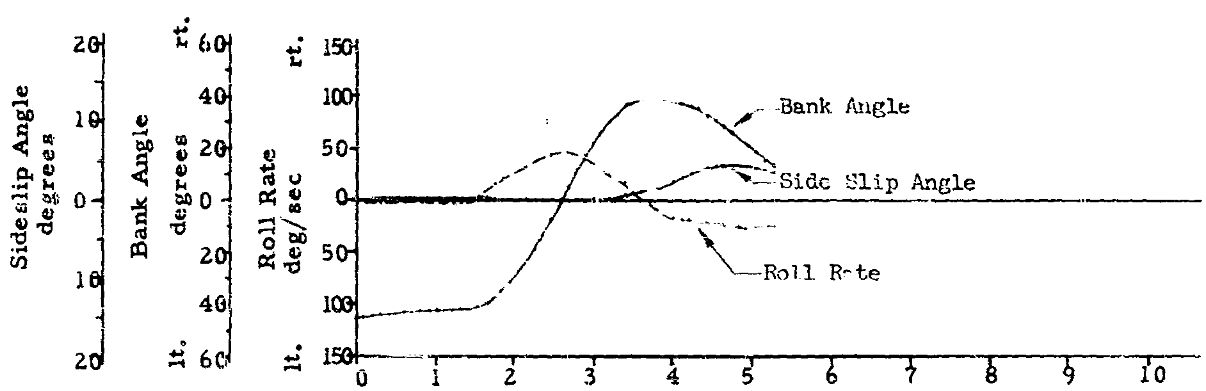
AILERON ROLLS

F-101A, USAF No. 53-2419
Power Approach Configuration

TRIM CONDITIONS

CAS	192	kts	CG pos.	32.2	%MAC
Alt.	16,020	ft	Weight	33,850	lb
Rudder pos.	0 Deg.		Aileron pos.	1.5 Deg. L.W.U.	
Stabilizer pos.	1.0° Nose Down		Rudder	Damper On	





APPENDIX I

Figure No. 134

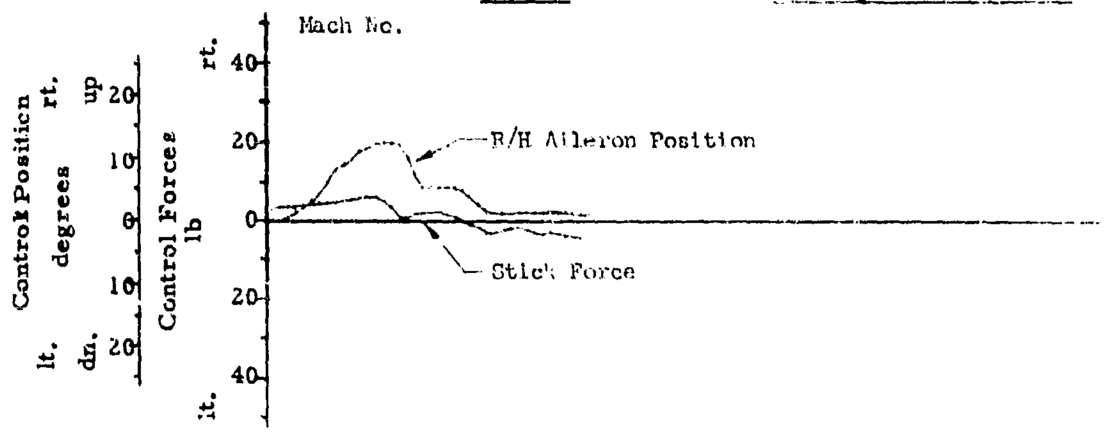
MAXIMUM ALLOWABLE DEFLECTION*

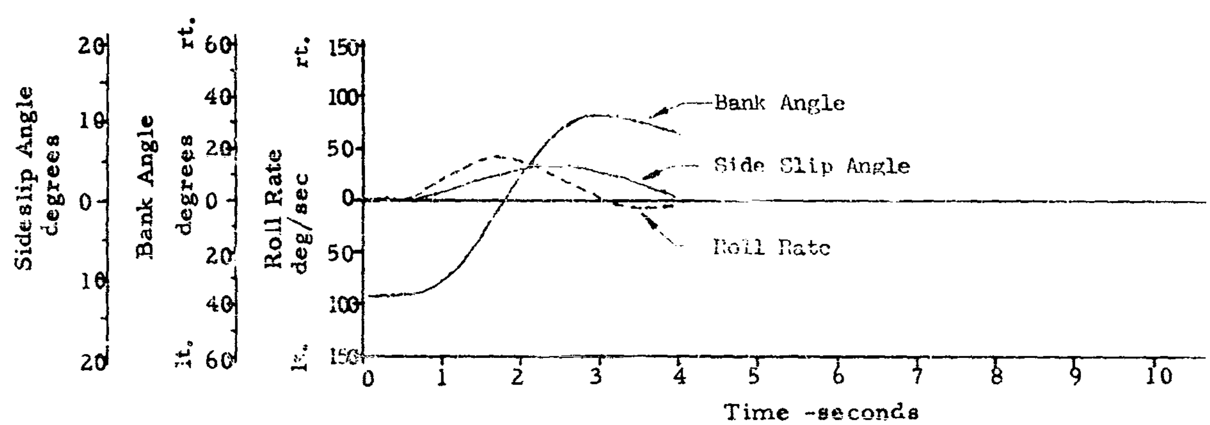
AILERON ROLLS

F-101A, USAF No. 53-2419
Power Approach Configuration

TRIM CONDITIONS

CAS	192	kts	CG pos.	32.2	%MA
Alt.	16,020	ft	Weight	33,850	lb
Rudder pos.	0 Deg.		Aileron pos.	1.8 Deg. L.W.U.	
Stabilizer pos.	1.0°Nose Down		Rudder	Damper Off	





APPENDIX I

Figure No. 135

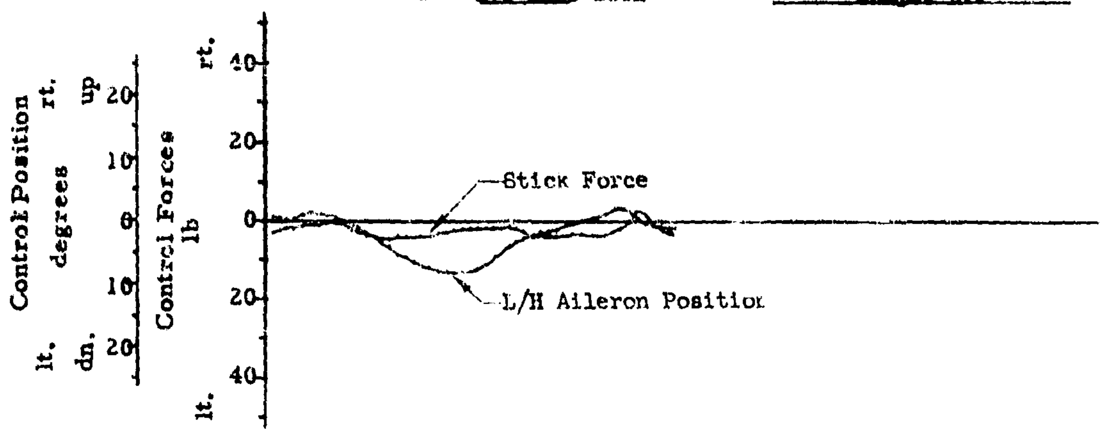
MAXIMUM ALLOWABLE DEFLECTION

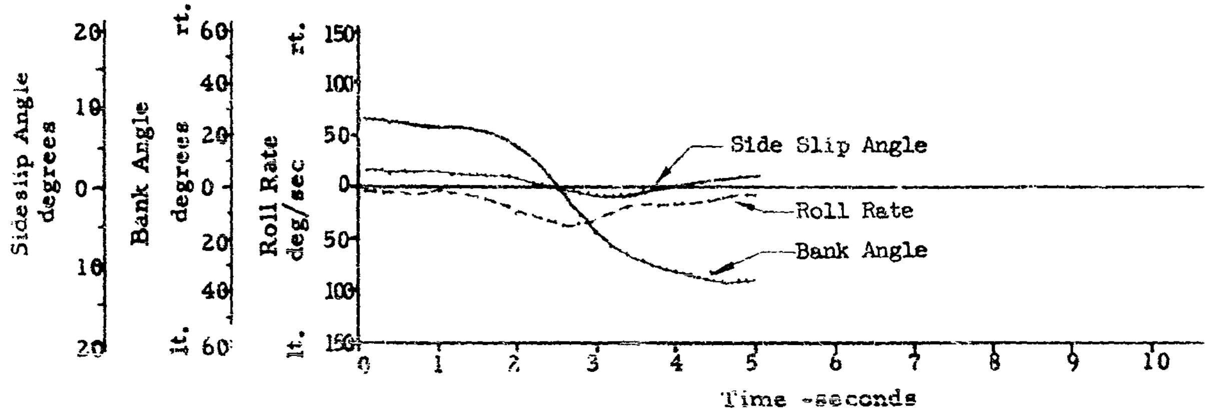
AILERON ROLLS

F-101A, USAF No. 53-2419
Power Approach Configuration

TRIM CONDITIONS

CAS 192 kts CG pos. 32.2 %MAC
Alt. 16,020 ft Weight 33,850 lb
Rudder pos. 0 Deg Aileron pos. 1.3 Deg. L.W.D.
Stabilizer pos. 1.0° Nose Down Rudder Damper Off





APPENDIX I

Figure No. 136

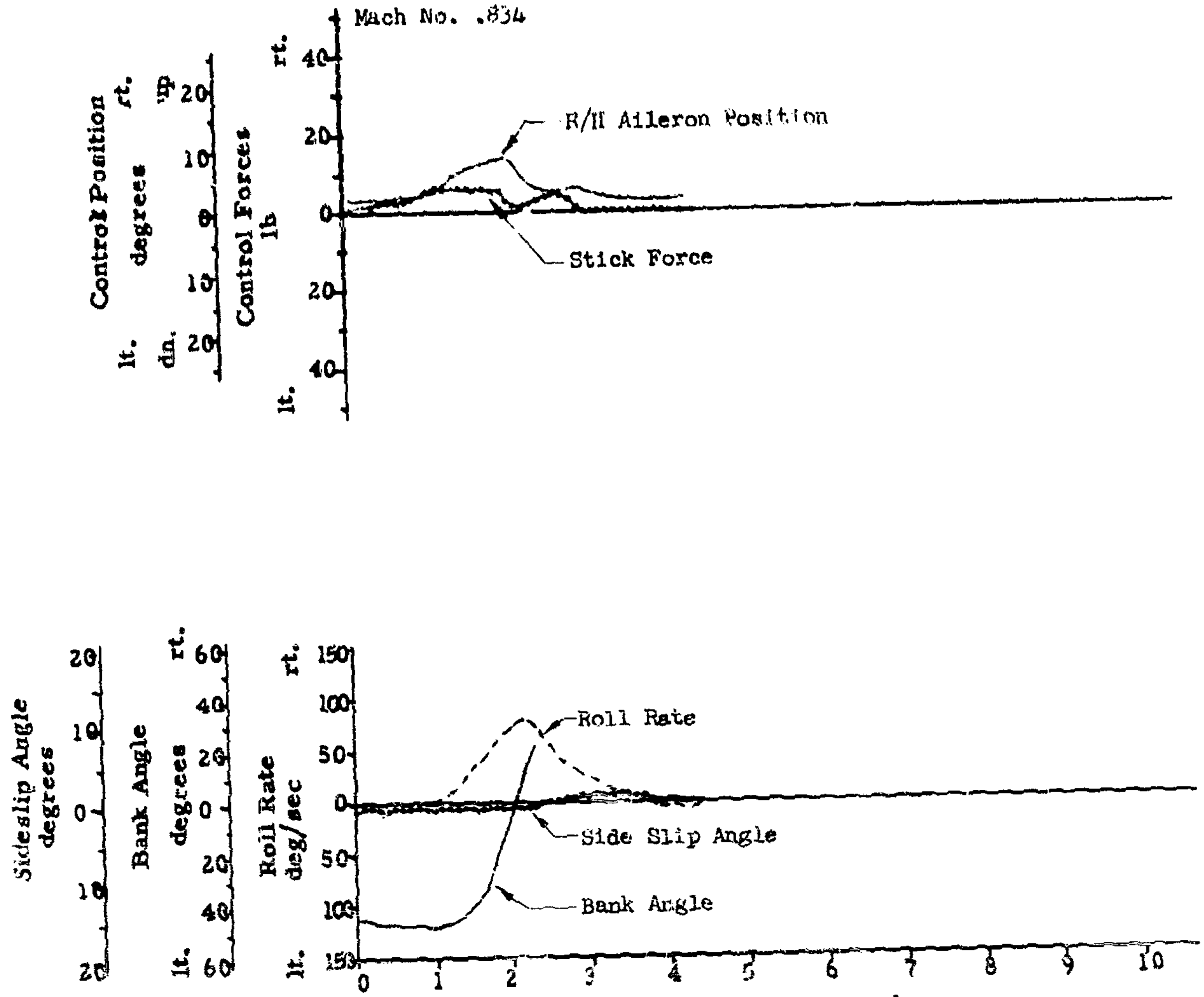
MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS 264.5 kts		CG pos. 34.3	%MAC
Alt. 38,150 ft		Weight 31,640	lb
Rudder pos. 0 Deg.		Aileron pos. 12 Deg. L.W.U.	
Stabilizer pos. 0 Deg.		Rudder Damper On	

Mach No. .834



APPENDIX I

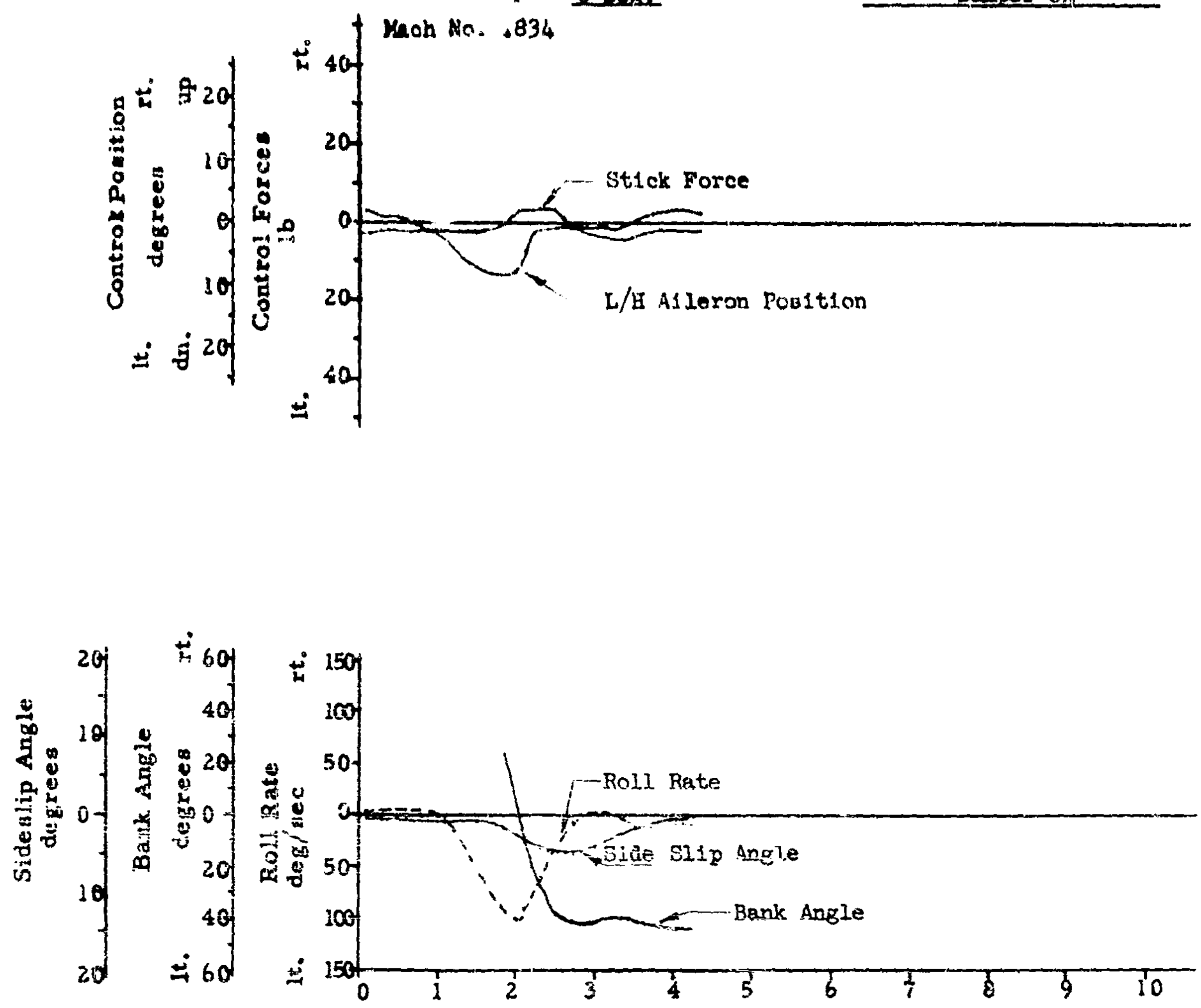
Figure No. 137

MAXIMUM ALLOWABLE DEFLECTION*

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	264.5	kts	CG pos.	34.3	%MAC
Alt.	38,150	ft	Weight	31,640	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	0 Deg.		Rudder	Damper On	



APPENDIX I

Figure No. 138

MAXIMUM ALLOWABLE DEFLECTION

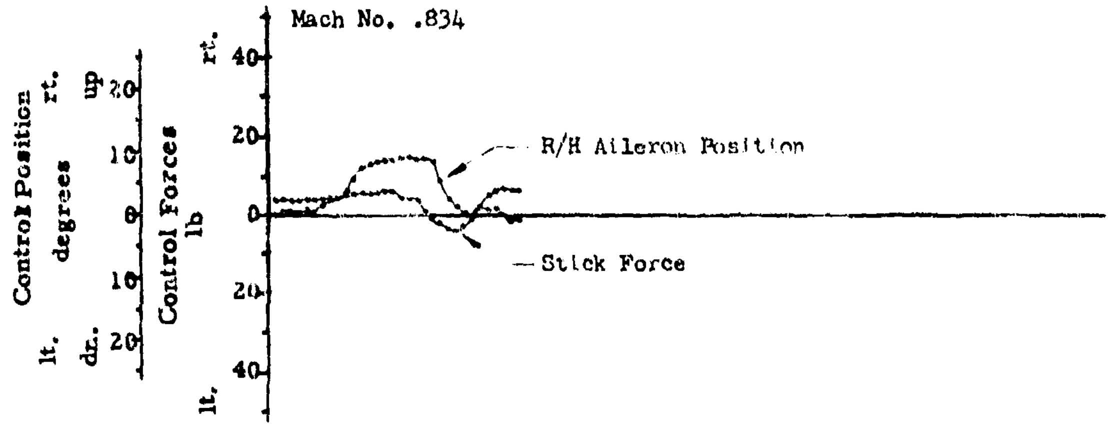
AILERON ROLLS

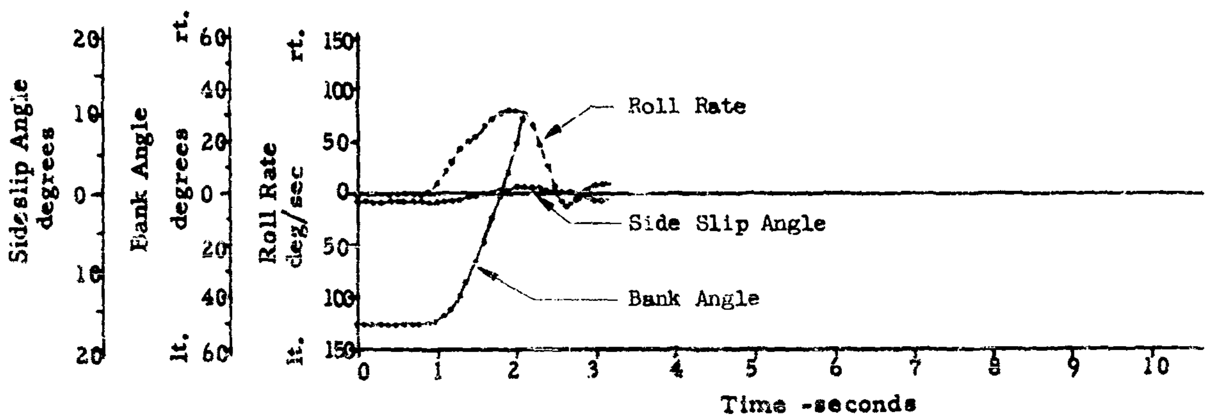
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	264.5	kts	CG pos.	34.3	%MAC
Alt.	38,150	ft	Weight	31,640	lb
Rudder pos.	0 Deg.		Aileron pos.	1.2 Deg. L.W.U.	
Stabilizer pos.	0 Deg.		Rudder	Damper Off	





APPENDIX I

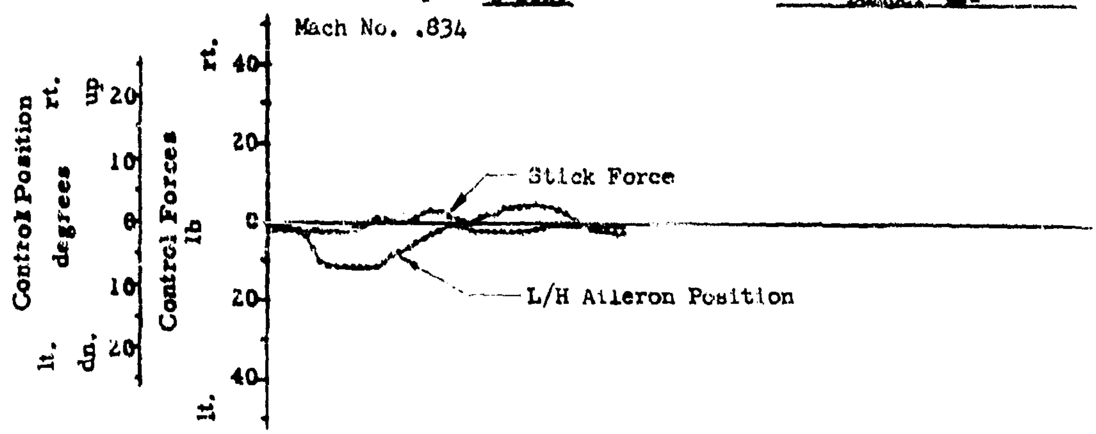
Figure No. 139

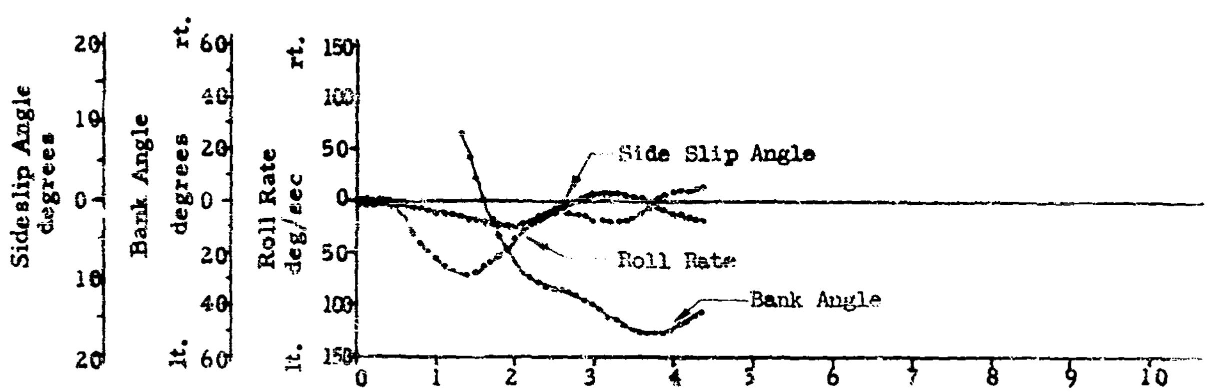
MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

F-101A, USAF No. 53-2419
Cruise Configuration
TRIM CONDITIONS

CAS	264.5	kts	CG pos.	34.3	%MAC
Alt.	38,150	ft	Weight	31,640	lb
Rudder pos.	0 Deg.		Aileron pos.	1.48 Deg. L.W.U.	
Stabilizer pos.	0 Deg.		Rudder	Damper Off	





APPENDIX I

Figure No. 140

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

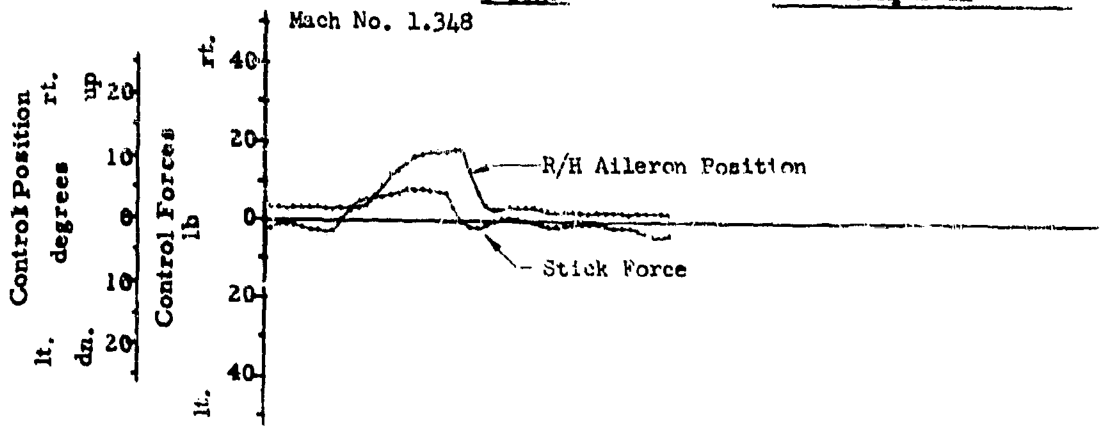
F-101A, USAF No. 53-2419

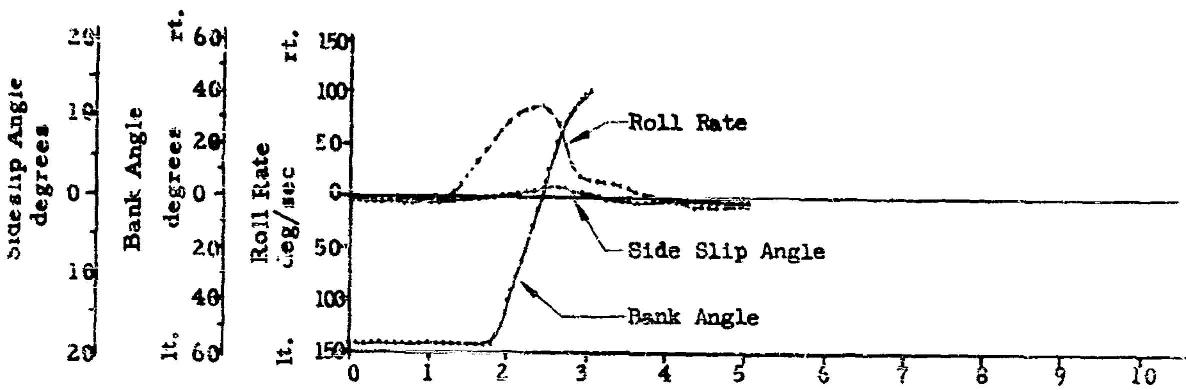
Combat Configuration

TRIM CONDITIONS

CAS	476	kts	CG pos.	33.6	%MAC
Alt.	36,360	ft	Weight	34,470	lb
Rudder pos.	0 Deg.		Aileron pos.	.6 Deg. L.W.U.	
Stabilizer pos.	0 Deg.		Rudder	Damper On	

Mach No. 1.348





Time -seconds

APPENDIX I

Figure No. 141

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

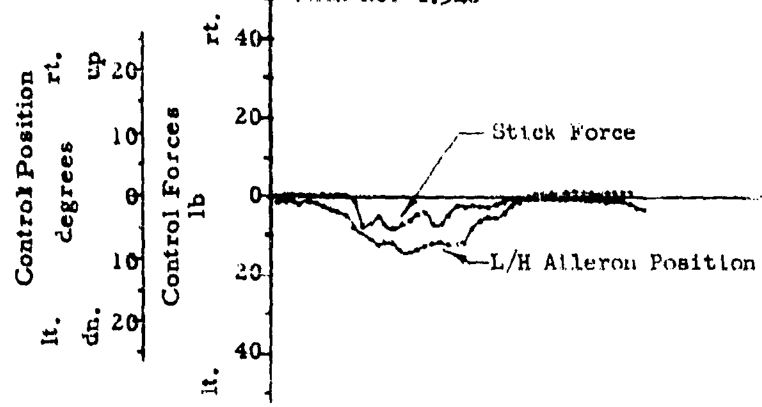
F-101A, USAF No. 53-2419

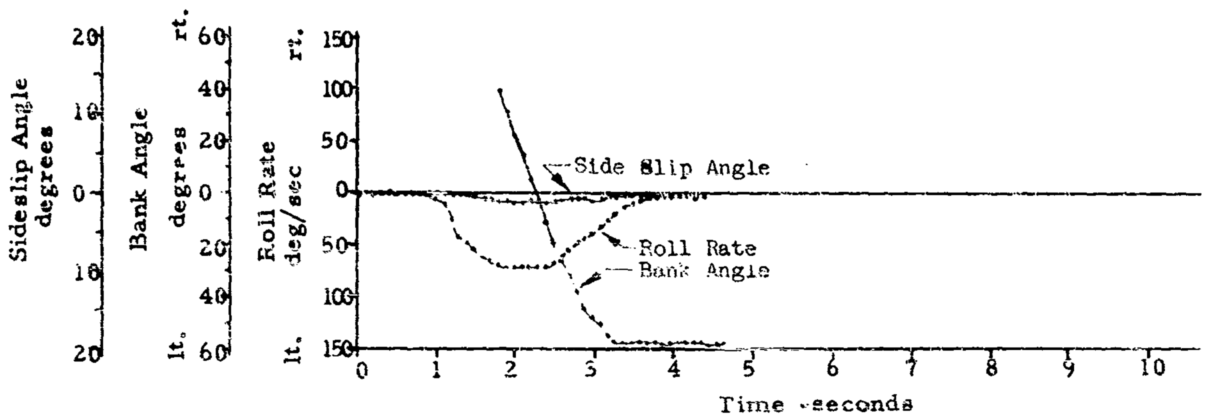
Combat Configuration

TRIM CONDITIONS

CAS	476	kts	CG pos.	33.6	%MAC
Alt.	36,360	ft	Weight	34,470	lb
Rudder pos.	0 Deg.		Aileron pos.	.6 Deg. L.W.U.	
Stabilizer pos.	0 Deg.		Rudder		

Mach No. 1.348





APPENDIX I

Figure No. 142

ADVERSE YAW

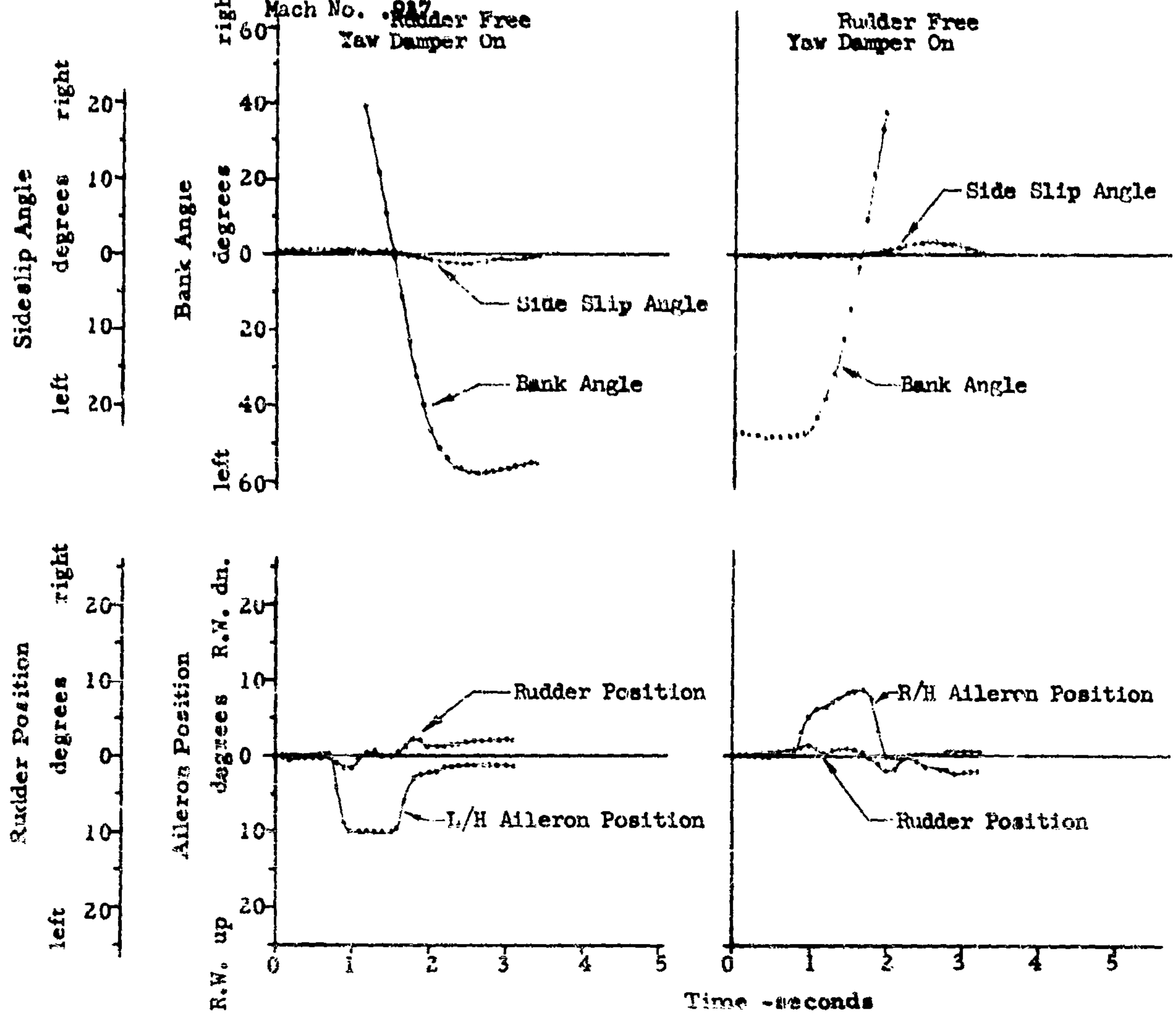
F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	326	kts	CG pos.	34.1	%MAC
Alt	35,370	feet	Weight	33,530	lb
Rudder Pos.	.5 Deg. Right		Aileron Pos.	0 Deg.	

Mach No. .917



APPENDIX I

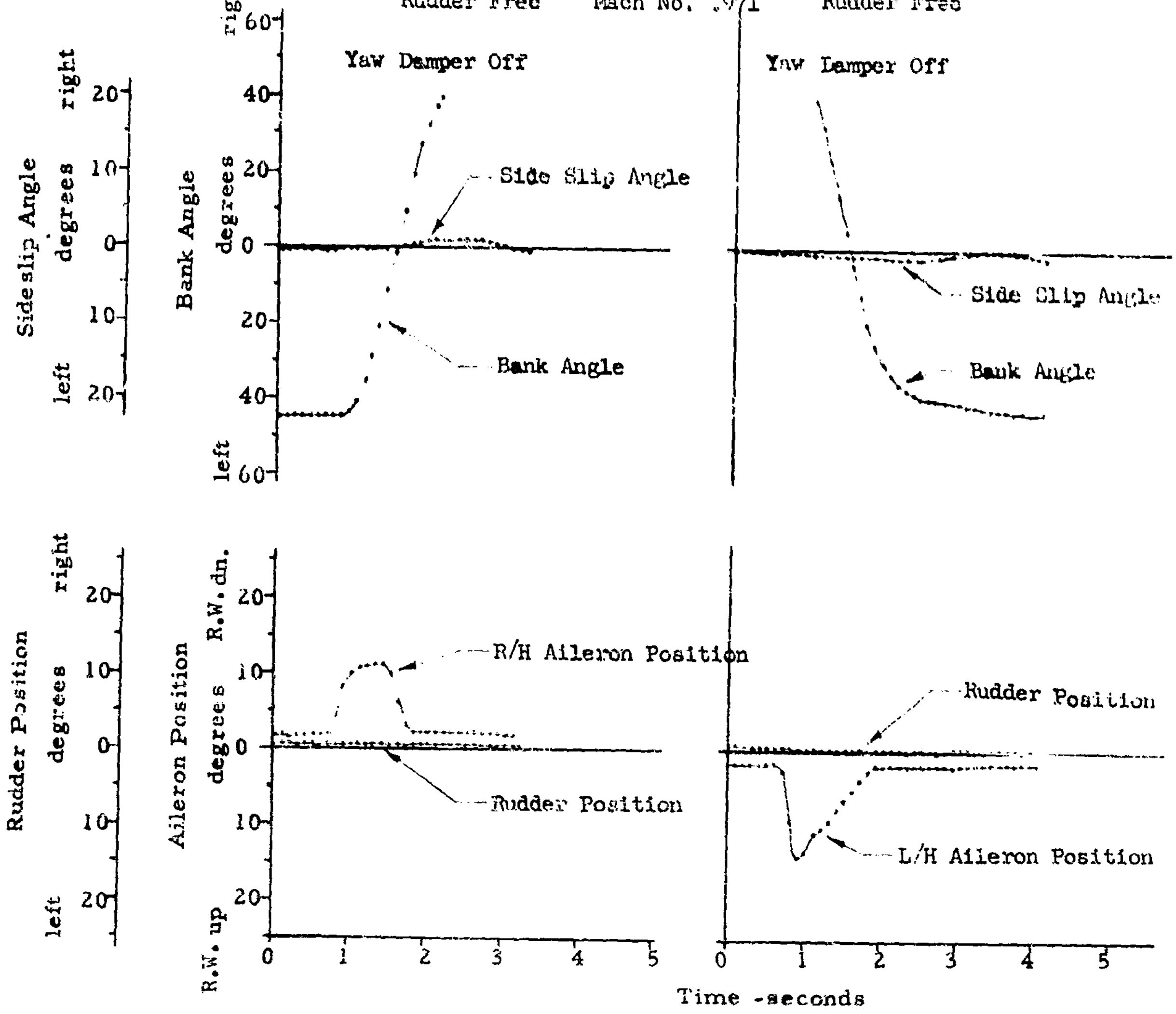
Figure No. 143

ADVERSE YAW

F-101A, USAF No. 53-2419
Cruise Configuration

TRIM CONDITIONS

CAS 326 kts CG pos. 34.1 %MAC
Alt 35,660 feet Weight 33,310 lb
Rudder Pos. .5 Deg. Right Aileron Pos. 0 Deg.
Rudder Free Mach No. .971 Rudder Free



APPENDIX I

Figure No. 144

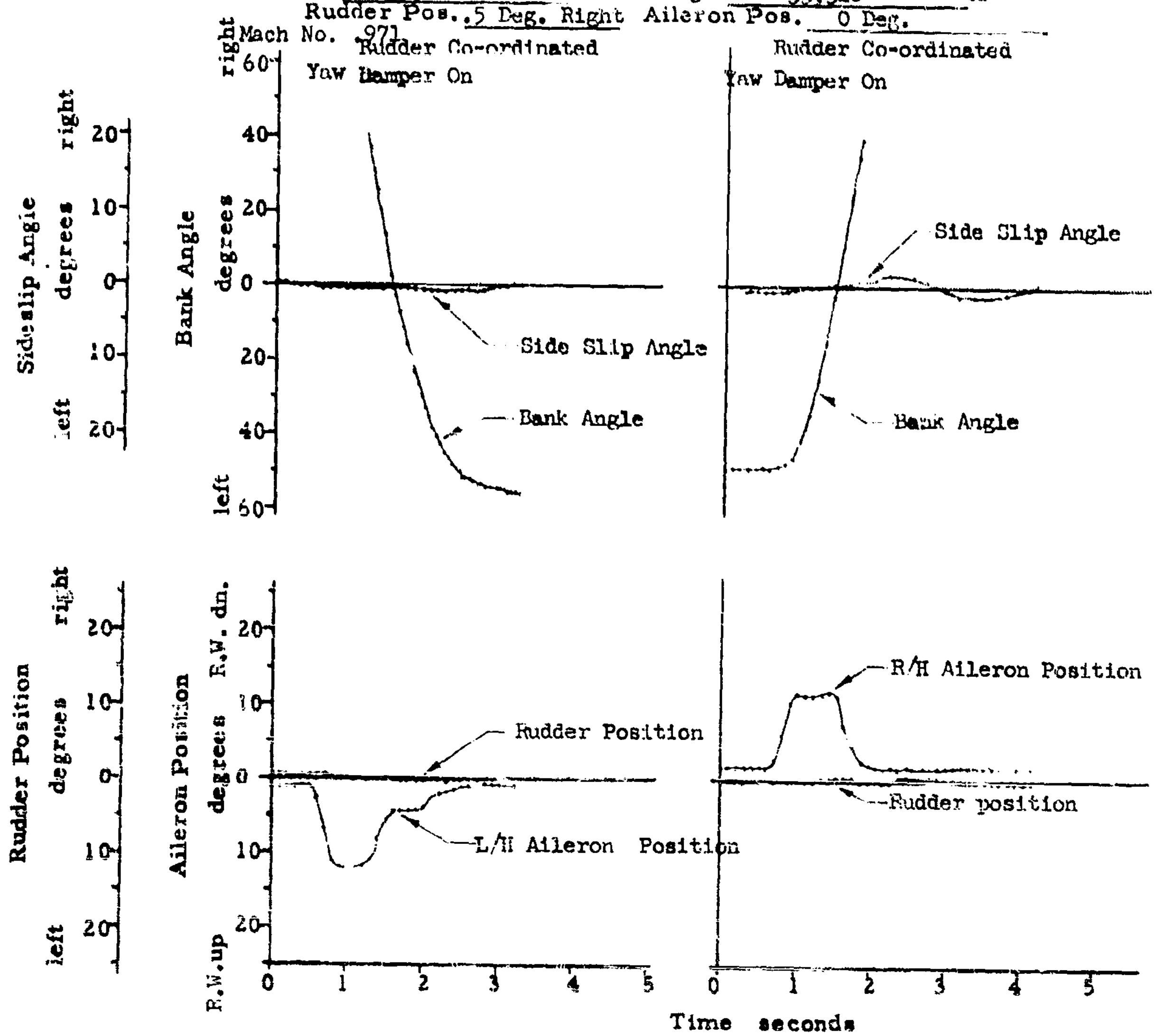
ADVERSE YAW

F-101A, USAF No. 53-2419

Cruise Configuration

TRIM CONDITIONS

CAS	326	kts	CG pos.	34.1	%MAC
Alt	35,660	feet	Weight	33,310	lb
Rudder Pos.	5 Deg. Right		Aileron Pos.	0 Deg.	
Mach No.	.971				



APPENDIX I

Figure No. 145

ADVERSE YAW

F-101A, USAF No. 53-2419

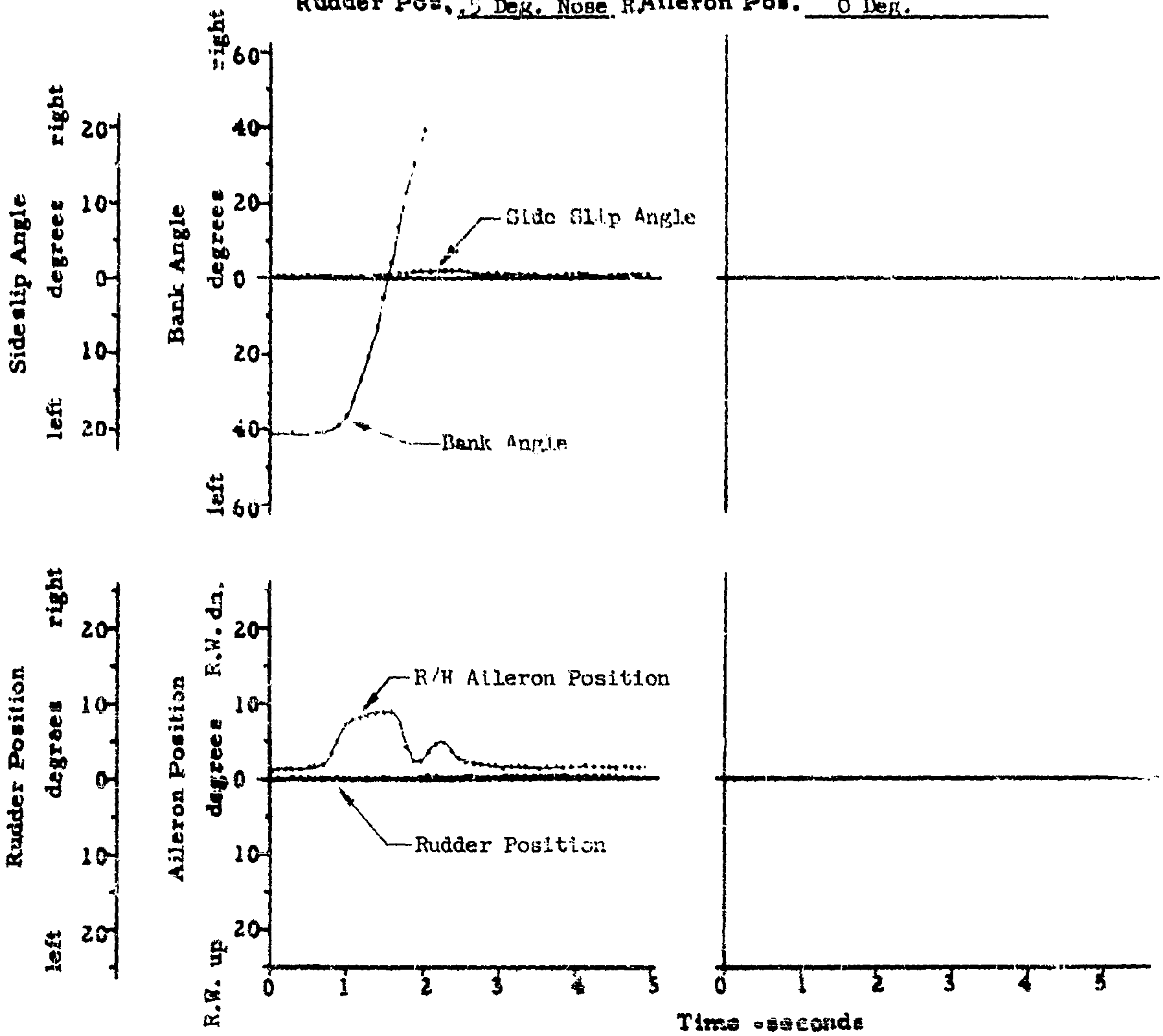
Combat Configuration

TRIM CONDITIONS

MACH 1.309 CG pos. 33.9 %MAC

Alt 35,020 feet Weight 32,050 lb

Rudder Pos. .5 Deg. Nose R Aileron Pos. 0 Deg.

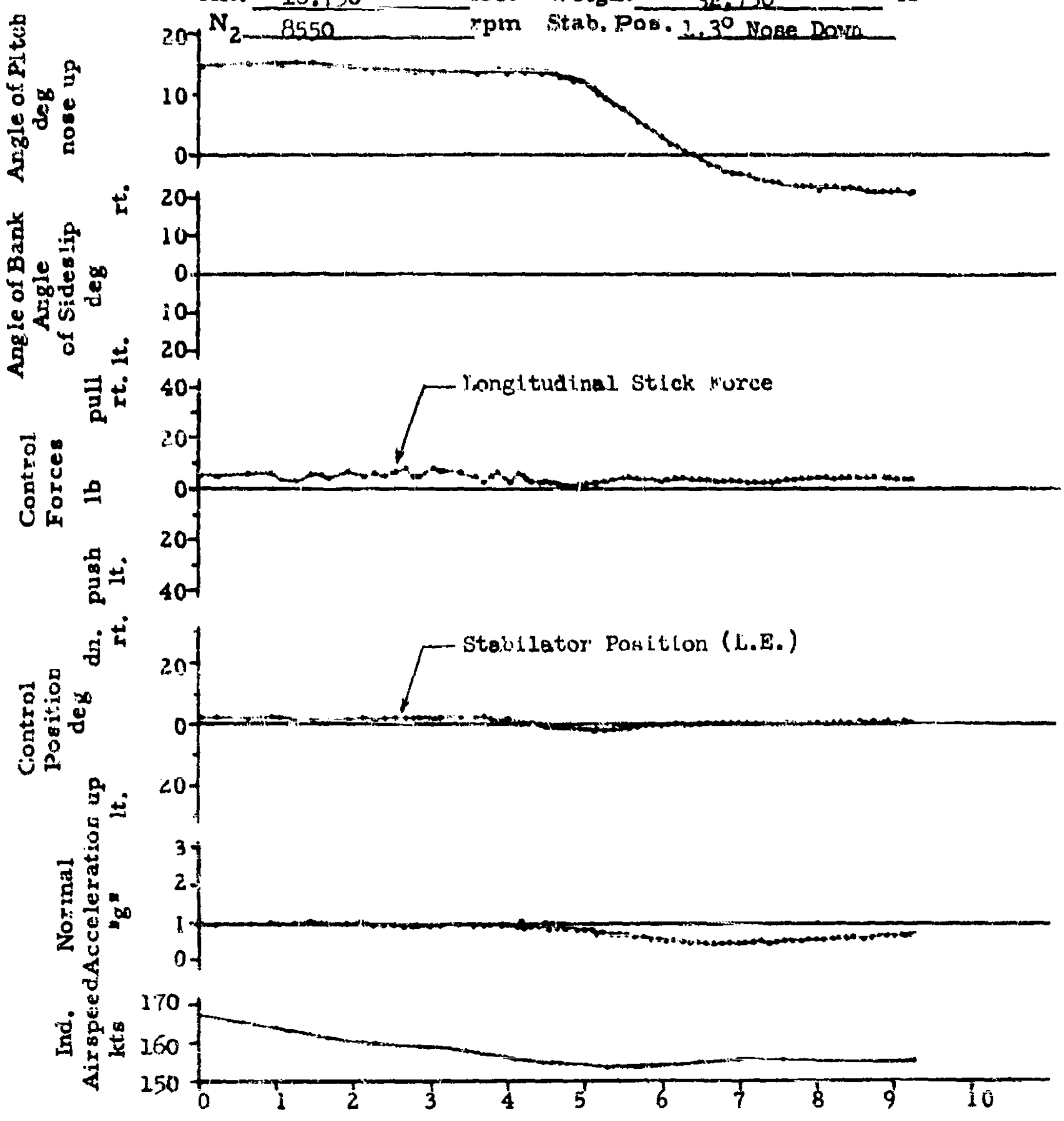


APPENDIX I

Figure No. 146
STALL TIME HISTORY
F-101A, USAF No. 53-2419
Configuration Cruise

TRIM CONDITIONS

CAS 415 kts CG pos. 34.0 %MAC
Alt. 16,750 feet Weight 32,750 lb
N_2 8550 rpm Stab. Pos. 1.3° Nose Down



APPENDIX I

Figure No. 147

STALL TIME HISTORY

F-101A, USAF No. 53-2419

Configuration Cruise

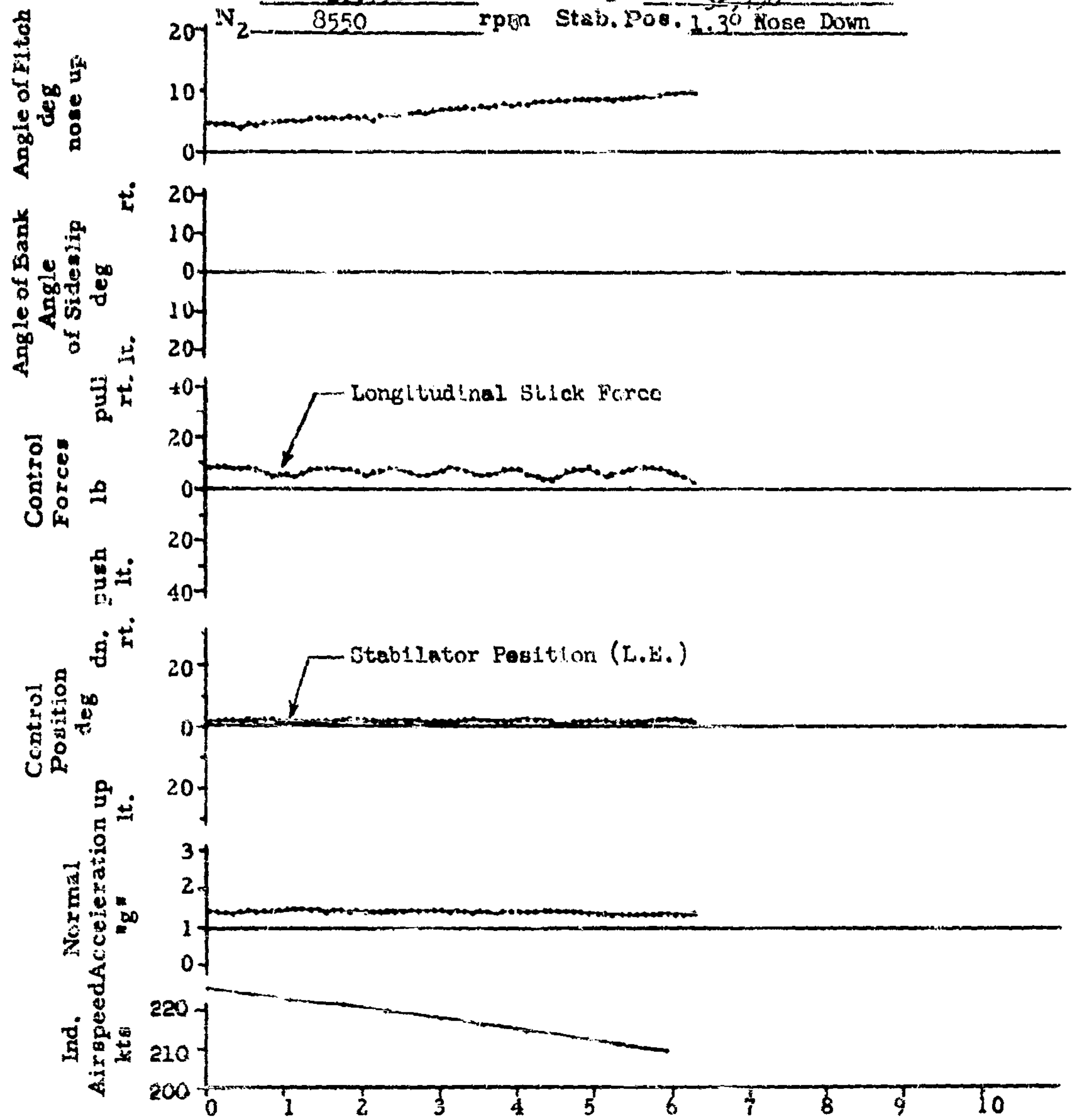
Accelerated Stall

TRIM CONDITIONS

CAS 415 kts CG pos. 34.0 %MAC

Alt. 16,750 feet Weight 32,750 lb

N_2 8550 rpm Stab. Pos. 1.3° Nose Down



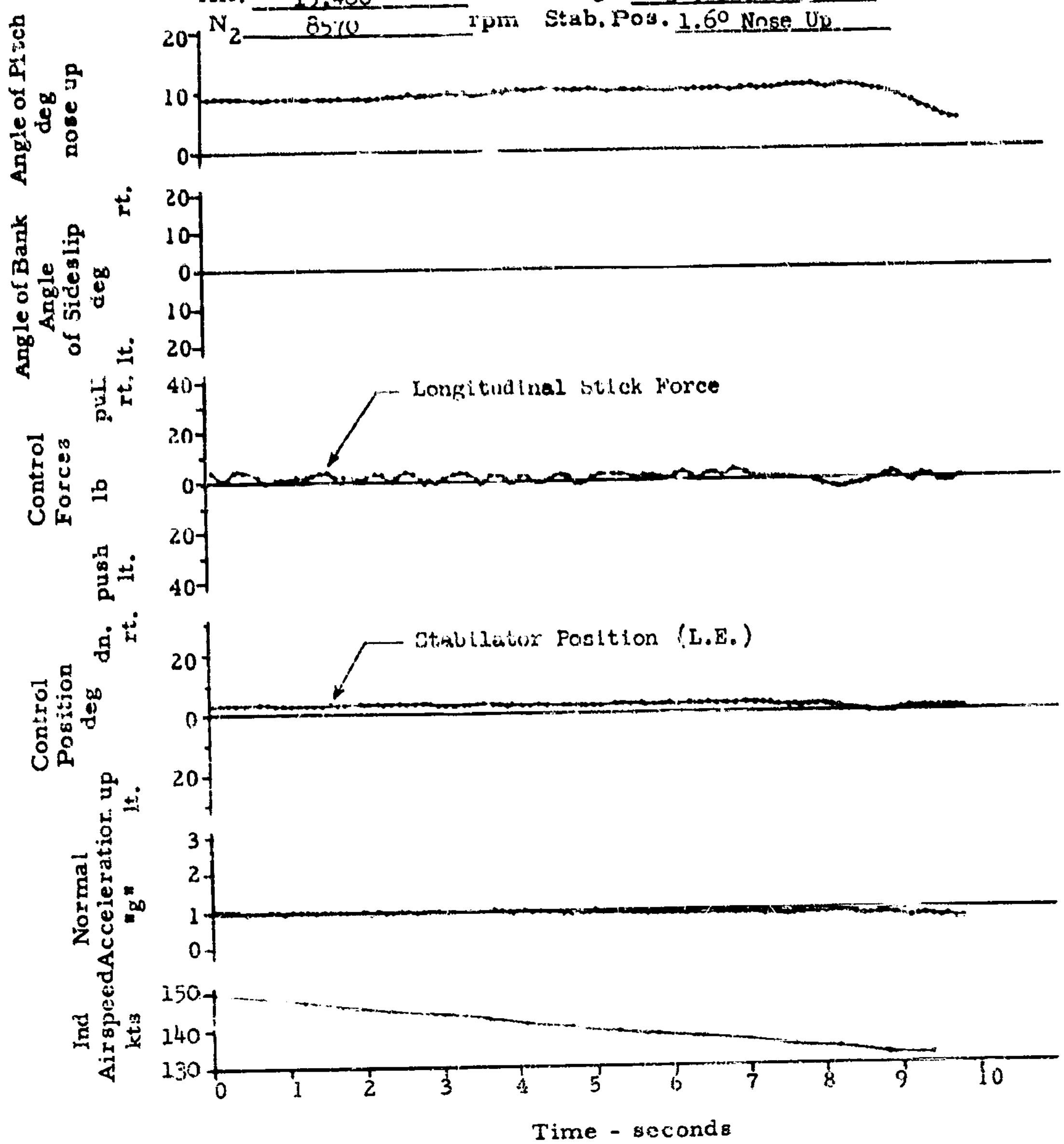
Time - seconds

APPENDIX I

Figure No. 148
STALL TIME HISTORY
F-101A, USAF No. 53-2419
Configuration Power Approach

TRIM CONDITIONS

CAS 214 kts CG pos. 29.5 %MAC
Alt. 15,400 feet Weight 32,250 lb
N_2 8570 rpm Stab. Pos. 1.6° Nose Up



APPENDIX I

Figure No. 149

STALL TIME HISTORY

F-101A, USAF No. 53-2419

Configuration Power Approach

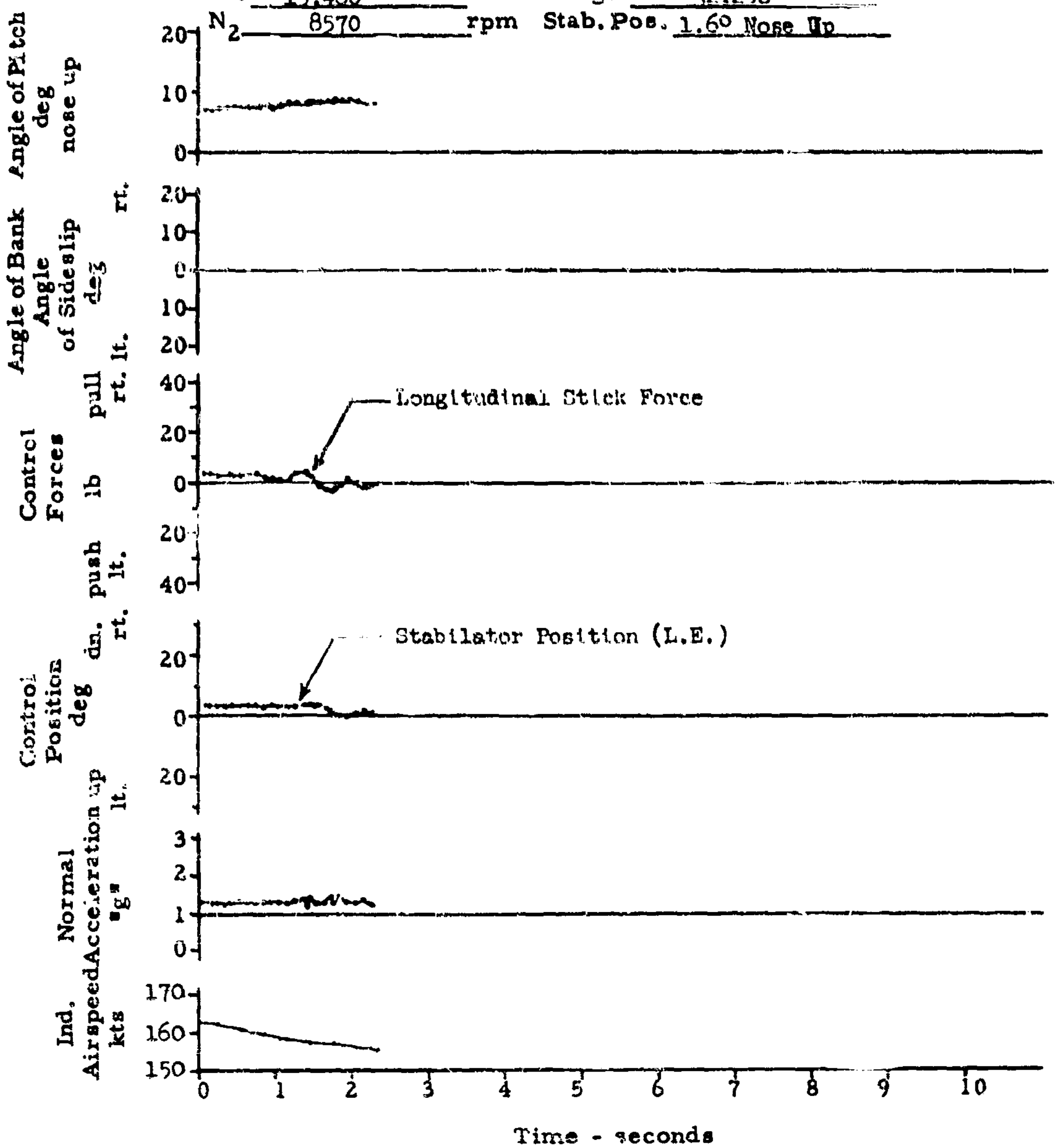
Accelerated Stall

TRIM CONDITIONS

CAS 214 kts CG pos. 29.5 %MAC

Alt. 15,480 feet Weight 32,250 lb

N_2 8570 rpm Stab. Pos. 1.6° Nose Up

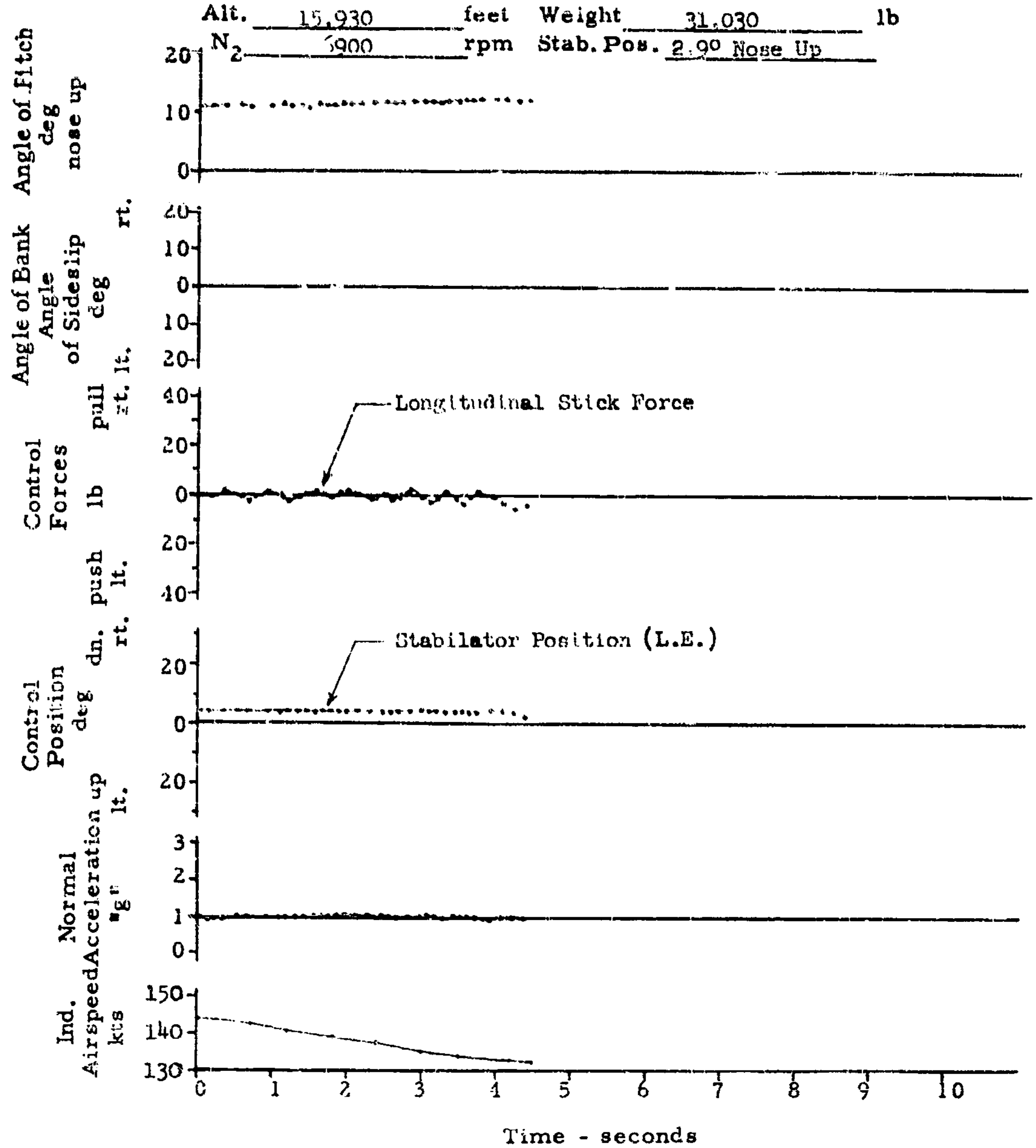


APPENDIX I

Figure No. 150
STALL TIME HISTORY
F-101A, USAF No. 53-2419
Configuration Landing

TRIM CONDITIONS

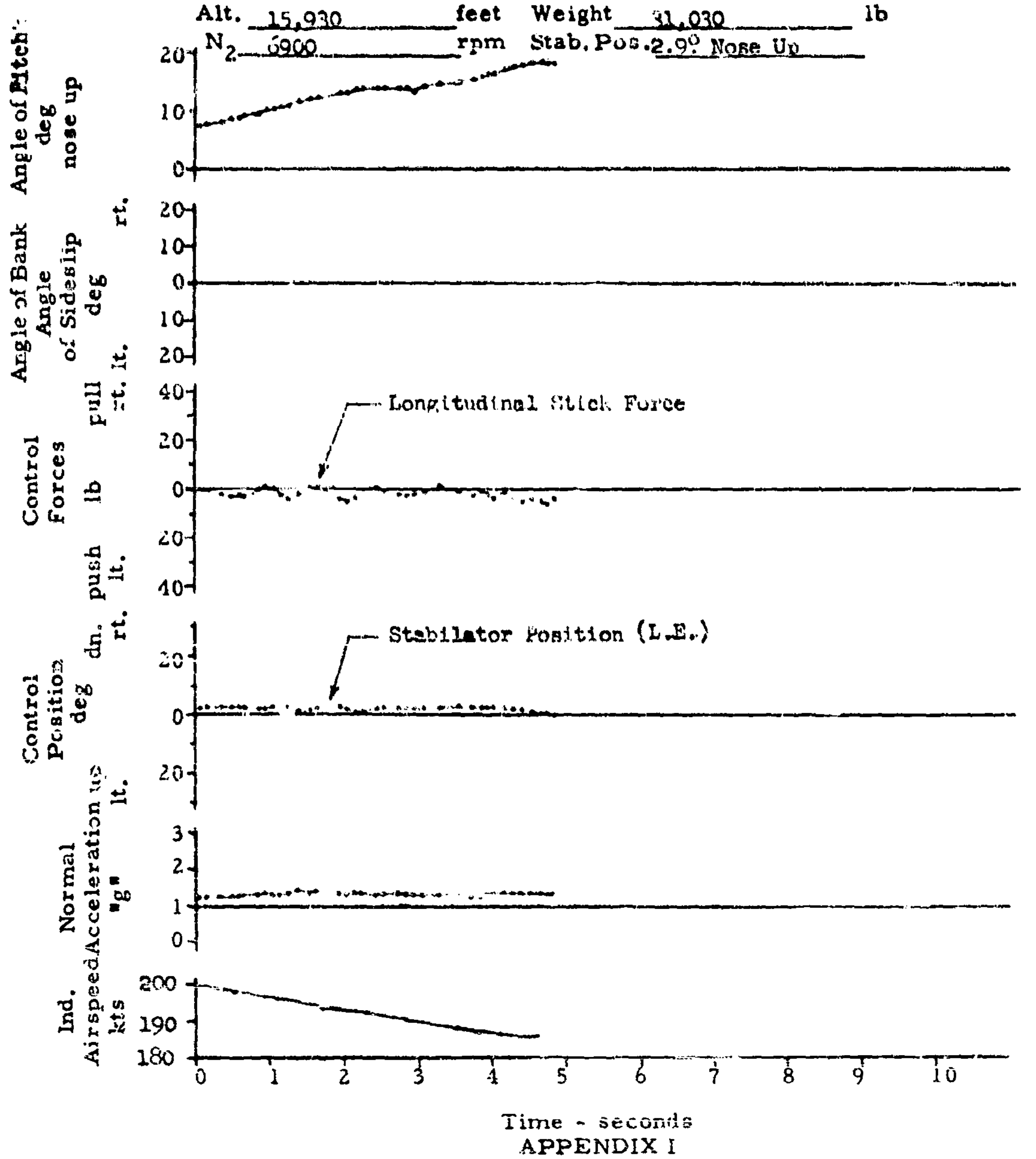
CAS 173 kts CG pos. 31.4 %MAC
Alt. 15,930 feet Weight 31,030 lb
N_2 5900 rpm Stab. Pos. 2.9° Nose Up



APPENDIX I

Figure No. 151
STALL TIME HISTORY
F-101A, USAF No. 53-2419
Configuration Landing
Accelerated Stall
TRIM CONDITIONS

CAS 173 kts CG pos. 31.4 %MAC
Alt. 15,930 feet Weight 31,030 lb
N_2 6900 rpm Stab. Pos. 2.9° Nose Up

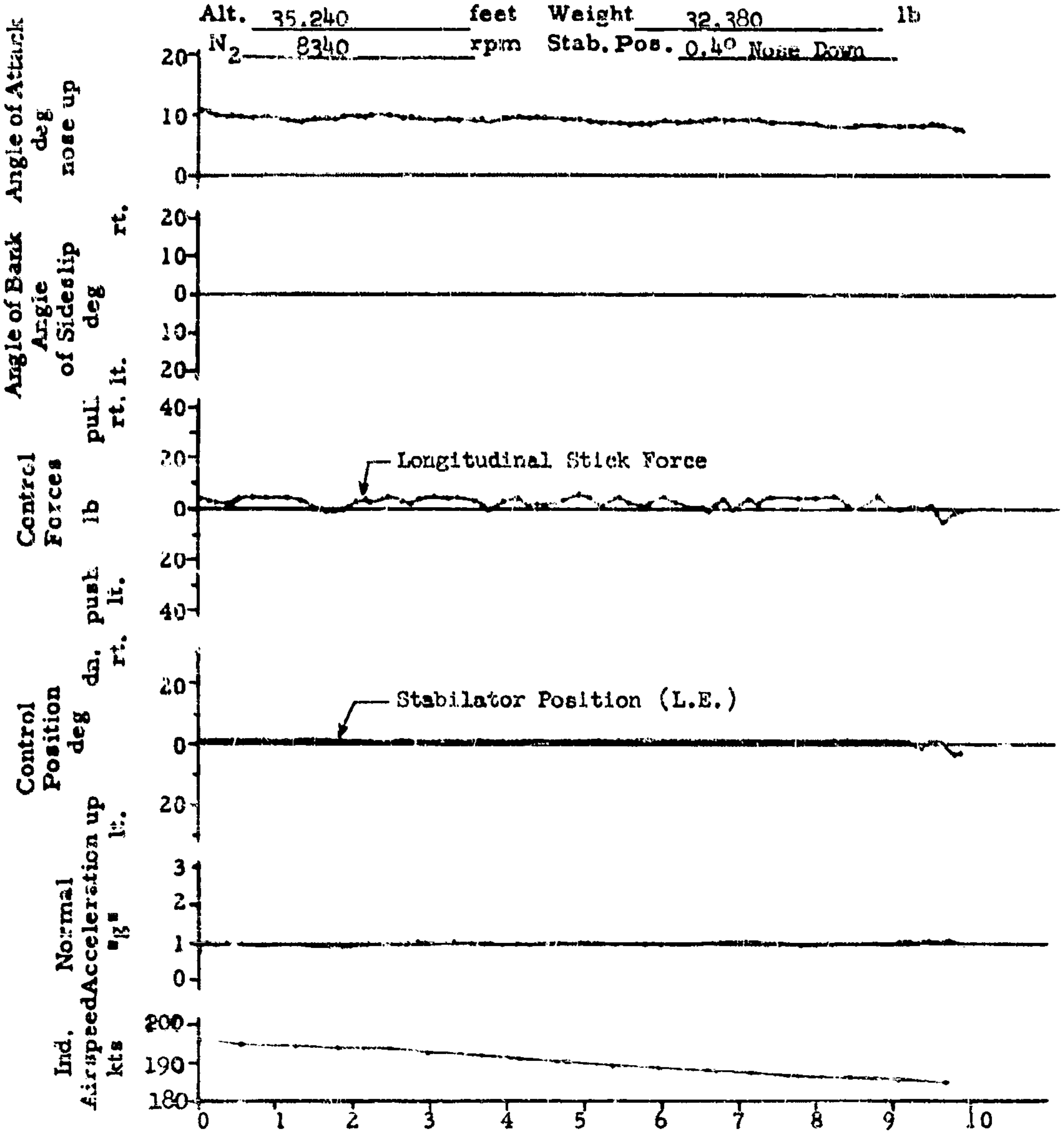


APPENDIX I

Figure No. 152
STALL TIME HISTORY
F-101A, USAF No. 53-2419
Configuration Cruise

TRIM CONDITIONS

CAS 201 kts CG pos. 32.4 %MAC
Alt. 35,240 feet Weight 32,380 lb
N_2 8340 rpm Stab. Pos. 0.4° Nose Down



Time - seconds
APPENDIX I

Figure No. 153

STALL TIME HISTORY

F-101A, USAF No. 53-2419

Configuration Cruise

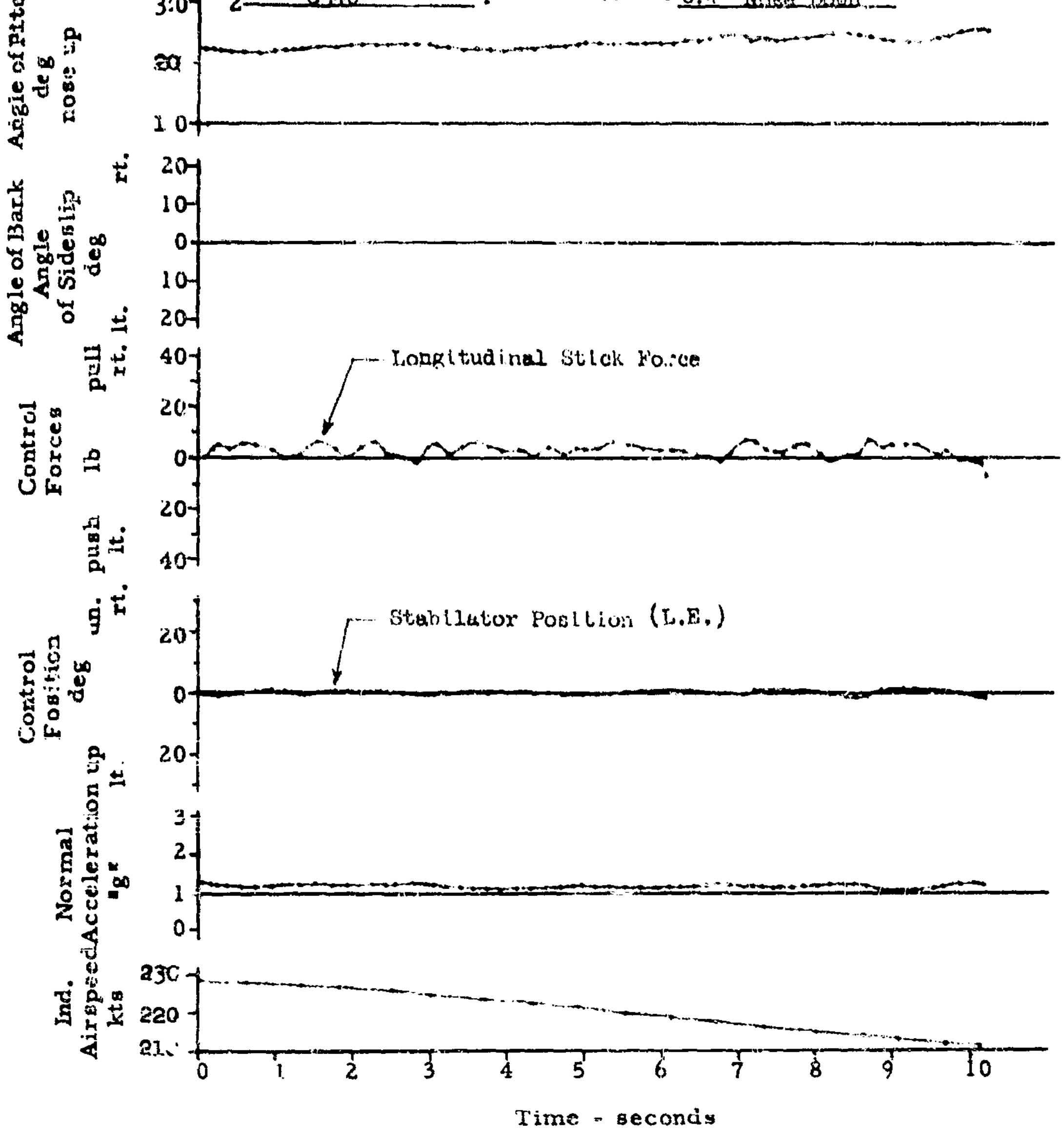
Accelerated Stall

TRIM CONDITIONS

CAS 201 kts CG pos. 32.4 %MAC

Alt. 35,240 feet Weight 32,380 lb

N_2 8340 rpm Stab. Pos. 0.40 Nose Down



APPENDIX I

APPENDIX II

TABLE OF CONTENTS

dimensions and general data

Design information and general dimensions of the airplane as listed in the McDonnell Aircraft Corporation detail specifications for the F-101A airplane are summarized as follows:

GENERAL DIMENSIONS

Span	39 ft 8.3 in
Length (overall)	67 ft 5.3 in
Height (overall)	18 ft

WING

Airfoil Section	
At root (theoretical)	
(centerline of aircraft)	NACA 65A007
At construction tip	
(theoretical extended section at tip)	NACA 65A006
Total Area	
Theoretical	367.99 sq ft
Actual	408.63 sq ft
Span, maximum	39.69 ft
Incidence	
At root	+1 deg
At construction tip	+1 deg
Dihedral	0 deg
Sweepback at 33.2% chord	35 deg
Aspect ratio	4.28
Mean aerodynamic chord	122.918 in
Aileron	
Span	86.0 in
Area aft of hinge line (one side) (theoretical)	14.74 sq ft
Deflection	
up	25 deg
down	25 deg
Chord (% wing chord)	28.5%
High Lift Device	
Type (trailing edge)	ZAP flap
Span, exclusive of cutouts	
(percent total wing span)	29.92%
Area (one side)	21.00 sq ft
Deflection	
up	0.0 deg
down	50 deg

SPEED BRAKE

Projected area (deflected 90 deg) one side	8.54 sq ft
Maximum deflection	50 deg

TAIL GROUP

Horizontal Tail Surfaces	
Span	15.5 ft
Total area	
Theoretical	75.08 sq ft
Actual	74.47 sq ft

Incidence (in neutral pos.) .... 0 deg
Sweepback (29.3% chord) .... 35 deg
Aspect ratio 3.30
Dihedral .... 10 deg
Horizontal stabilizer movement
(with respect to W.L.)
L.E. up .... 16 deg
L.E. down .... 20 deg
Vertical Tail Surfaces
Span .... 7.5 ft
Area (theoretical) (above W.L. 92.00) .... 84.88 sq ft
Sweepback (59.1% chord) .... 35 deg
Aspect ratio 0.663

BODY GROUP

Fuselage
Length .... 804.11 in.
Width (max.) .... 92.00 in.
Height (max.) .... 84.24 in.

LANDING GEAR

Main landing gear
casing size .... 32 x 8.8
tread .... 19 ft 10.7 in.
oleo stroke .... 12 in.
Nose Gear
casing size .... 18 x 5.5
strut stroke .... 9 in.

FUEL CAPACITY (Internal)

No. 1 tank, fuselage non-self-sealing .... 794 gal
No. 2 tank, fuselage self-sealing .... 439 gal
(+ 13 when IFR)
No. 3 tank, fuselage non-self-sealing .... 245 gal
No. 4 tank, fuselage non-self-sealing .... 349 gal
No. 5 tank, fuselage non-self-sealing .... 319 gal
Total internal 2146 gal

■ operational limitations

GROSS WEIGHT

Maximum Design Gross Weight .... 37,000 lb

LIMIT FLIGHT LOAD FACTORS (Maneuver)

Design Gross Weight .... 37,000 lb
positive .... +7.33
negative .... −3.0
Maximum Alternate Gross Weight .... 46,000 lb
positive .... +5.6
negative .... −2.1
Minimum Flying Weight .... 27,500 lb
positive .... +7.33
negative .... −3.0

Limit Gust Load Factors
Design Gross Weight .... 37,000 lb
positive .... +3.06
negative .... −1.06
Maximum Alternate Gross Weight .... 46,000 lb
positive .... +2.66
negative .... −0.66
Minimum Flying Weight .... 27,500 lb
positive .... +3.77
negative .... −1.77

LIMIT GROUND LOAD FACTORS

Taxi Load Factor at Maximum
Alternate Gross Weight (46,000 lb) 2.0

LIMIT DIVING SPEED

Design Gross Weight
At 3,500 ft altitude .... 760 knots EAS
At 35,000 ft altitude .... 502 knots EAS
At 50,000 ft altitude .... 260 knots EAS

Limit diving speed for the maximum alternate gross weight shall be at least the speed corresponding to maximum level flight speed at that weight and the speed shall be as much higher than the maximum level flight speed as possible without major modification to the airplane structure when designed for the design gross weight.

LIMIT STALLING SPEED (31,000 lb)

Without trailing edge flaps, from sea level to
12,000 feet .... 153 knots EAS
With trailing edge flaps and gear down, from
sea level to 12,000 feet .... 44 knots EAS

LIMIT OPERATING SPEEDS — 12,000 feet

Trailing edge flap
extension to 50 deg .... 230 knots EAS
retraction from 50 deg .... 250 knots EAS*

* A pressure operated switch insures flap retraction at this speed.

Canopy Opening and Closing .... 90 knots EAS ultimate
Alighting gear extension and retraction .... 250 knots EAS
Refueling probe extension and retraction .... 300 knots EAS

The following limitations based on flight operating boundaries which had been investigated during Phase I were placed on the F-101A airplane by the contractor. These are:

SPEED

1.55 Mach number or 650 knots EAS, whichever is lower

NORMAL LOAD FACTOR

Symmetrical Flight Maneuvers
Upper limit .... +4 to 1.5 Mach no.
+1 to 1.55 Mach no.
Lower limit* .... +0.5

The lower limit for normal acceleration was 0.5 g because of trouble experienced with the constant speed transmissions which drive the AC generators. These have been found to be very sensitive to oil supply interruption.

ROLLING MANEUVERS

Aileron Rolls

Angle of bank . . . . 45°

Aileron deflection . . . . 10° up to 500 kts IAS
5° over 500 kts IAS

The limitation on roll angle was necessary because of the inertia coupling problem on modern fighters which the contractor had not yet been able to investigate. The limitation on aileron angle at high speeds is necessary until completion of static tests.

AFTERBURNER OPERATION

Supersonic Speeds . . . . 10 min

Subsonic Speeds . . . . 15 min (max. allowed by engine manufacturer)

CABIN PRESSURIZATION

Differential pressure . . . . 2.75 psi, at all times

FUEL VENT SYSTEM

Do not exceed 2 psi fuel tank pressure

■ power plant

Number: two

Type: turbo-jet

Manufacturer: Pratt and Whitney Company

Model: USAF, model XJ57-P-13

Rating: The following table shows the thrust rating of the uninstalled engine under standard sea level static conditions:

Maximum (afterburner operating) . . . . 14,000 lb

Military (afterburner not operating) . . . . 9,220 lb

Normal (afterburner not operating) . . . . 8,800 lb

■ weight and balance

The following weight and balance was obtained by weighing the aircraft in a closed hangar with full fuel and oil and gear down. The CG location will move forward approximately 0.2% with gear up.

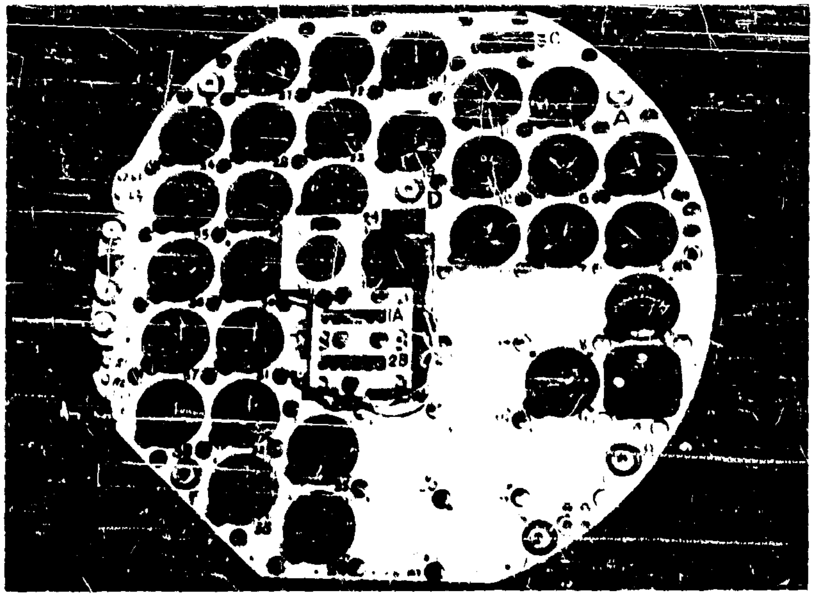
Basic airplane*	Wt. — lb	ARM — in.	Mom/1000
	27,330	499.4	13,647.59
Fuel at 6.54 lb/gal Tank no.			
1	5,000	379.0	1,895.00
2	2,685	445.6	1,196.44
3	1,455	509.8	741.76
4	1,960	572.0	1,121.12
5	1,970	645.6	1,271.83
Basic wt. as weighed:	40,400	491.9	19,873.74

* Basic airplane includes test instrumentation and 9 gals of oil (67.5 lb)

The weight and balance for the complete flight test program are as follows:

Flt. No.	Basic wt. lb	Crew lb	Engine Start Gr. Wt. -- lb	Approx. CG % MAC
1	27,330	200	40,900	30.73
2	27,330	200	39,160	30.62
3	27,330	200	40,440	31.48
4	27,330	200	40,090	30.34
5	27,330	200	40,305	29.83
6	27,330	200	40,085	30.75
7	27,330	200	40,270	31.07
8	27,330	200	40,755	31.00
9	27,330	200	40,120	30.82
10	27,330	200	40,605	30.83
11	27,330	200	40,755	31.15
12	27,330	200	40,770	31.23
13	27,330	200	40,775	30.99
14	27,330	200	40,565	31.97
15	27,330	200	31,020	35.63
16	27,330	200	40,690	31.07
17	27,330	200	40,460	31.44
18	27,330	200	40,645	31.15
19	27,330	200	40,840	31.15
20	27,330	200	40,655	31.07
24	27,330	200	40,710	31.00
32	27,330	200	40,680	30.58

The basic weight includes test instrumentation and 9 gallons of oil.
The approximate cg is with landing gear down.



Instrumentation

Instrumentation for performance and stability flight testing were installed by the contractor as per AFFTC specification with approved modifications or deletions. All instruments were calibrated by the AFFTC Instrumentation Branch. The following test instrumentation was used for the Phase II flight tests.

Pilot's Cockpit Instrumentation

1. Correlation counter
2. Photo-panel and oscillograph control
3. Altimeter (nose boom)
4. Airspeed (nose boom)
5. Machmeter (nose boom)
6. Tachometer-high pressure compressor (L/H engine)
7. Tachometer-high pressure compressor (R/H engine)
8. Turbine out pressure (dual indicator), P_{t_7}
9. Free air temperature
10. Fuel quantity gage (total)
11. Angle of sideslip
12. Accelerometer (direct reading)
13. Exhaust gas temperature (L/H and R/H engine)

Photo-panel (35 mm photo recorder)

1. Correlation counter (photo recorder)
2. Altimeter (nose boom)
3. Airspeed (nose boom)
4. Altimeter (ship's system)
5. Airspeed (ship's system)
6. Machmeter (nose boom)
7. Free air temperature gage
8. Photo-observer temperature gage
9. 8-day clock
10. 10-second stopwatch
11. RPM-low pressure rotor, L/H
12. RPM-low pressure rotor, R/H
13. RPM-high pressure rotor, L/H
14. RPM-high pressure rotor, R/H
15. Compressor inlet pressure, L/H (P_{t_2})
16. Turbine out total pressure, L/H (P_{t_7})
17. Turbine out total pressure, R/H (P_{t_7})
18. Turbine out temperature, L/H (t_{t_7})
19. Turbine out temperature, R/H (t_{t_7})
20. Stabilator position
21. Rudder position
22. Left aileron position
23. Right aileron position
24. Fuel quantity gage (tank no. 1)
25. Fuel quantity gage (tank no. 2)
26. Fuel quantity gage (tank no. 3)
27. Fuel quantity gage (tank no. 4)
28. Fuel quantity gage (tank no. 5)
29. Engine fuel inlet pressure, L/H
30. Engine fuel inlet pressure, R/H
31. Afterburner fuel pressure, L/H (manifold)
32. Afterburner fuel pressure, R/H (manifold)
33. Accelerometer (direct reading)
34. Speed brake position (dual indicator)
35. L/H Oscillograph counter number
36. R/H Oscillograph counter number

Light Indications

37. Main landing gear lift-off signal light
38. Nose gear lift-off signal light
39. Speed brake extension signal light
40. Inter-compressor bleed door, L/H, signal light
41. Inter-compressor bleed door, R/H, signal light
42. R/H Oscillograph correlation light
43. L/H Oscillograph correlation light
44. Brown recorder correlation light

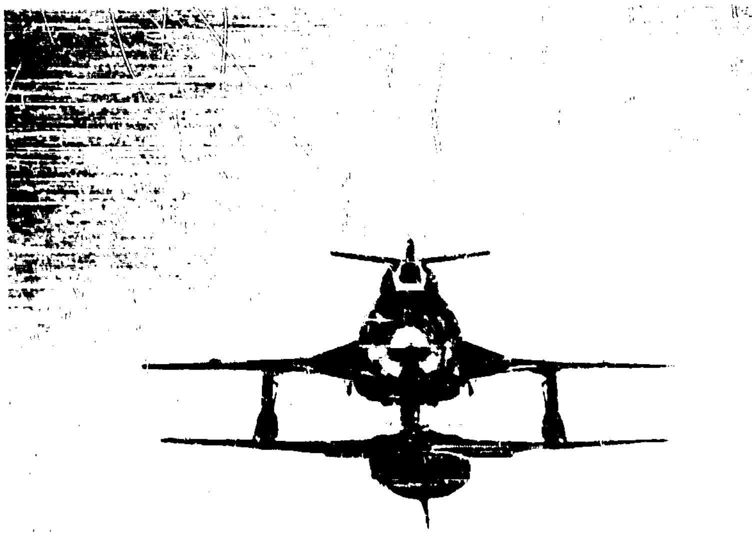
L/H Oscillograph (18 channel)

1. Free stream dynamic pressure (nose boom), airspeed
2. Static pressure (nose boom), altitude
3. Longitudinal stick force
4. Stabilator position
5. Rudder pedal force, R/H
6. Rudder pedal force, L/H
7. Rudder position (lower)
8. lateral stick force (aileron)
9. Left aileron position
10. Right aileron position (out board)
11. Normal acceleration (at cg)
12. Angle of attack
13. Angle of sideslip
14. Rate of roll
15. Rate of yaw
16. Rate of pitch
17. Angle of pitch
18. Angle of roll
19. Oscillograph counter no.

R/H Oscillograph (18 channel)

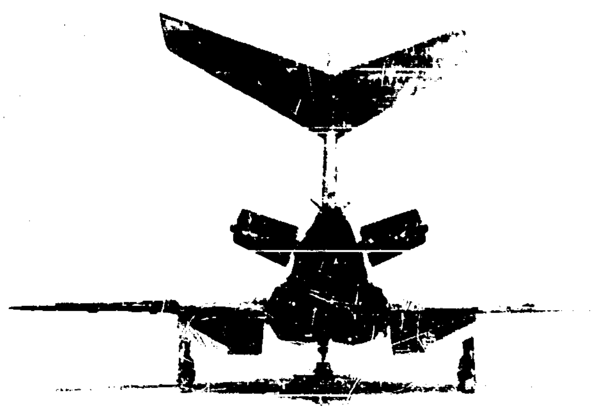
1. Engine fuel flow, L/H
2. Engine fuel flow, R/H
3. Afterburner fuel flow, L/H
4. Afterburner fuel flow, R/H
5. Free air temperature, McDonnel probe
6. Compressor inlet temperature, L/H (McDonnel outer temperature probe)
7. Afterburner nozzle position, L/H
8. Afterburner nozzle position, R/H

■ Identification photos

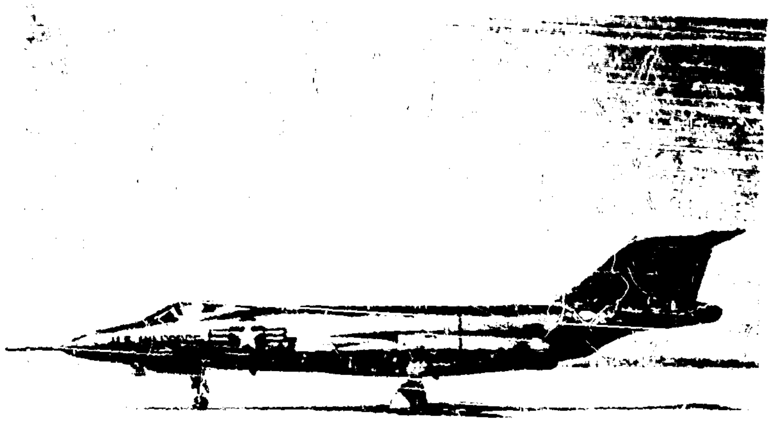


front

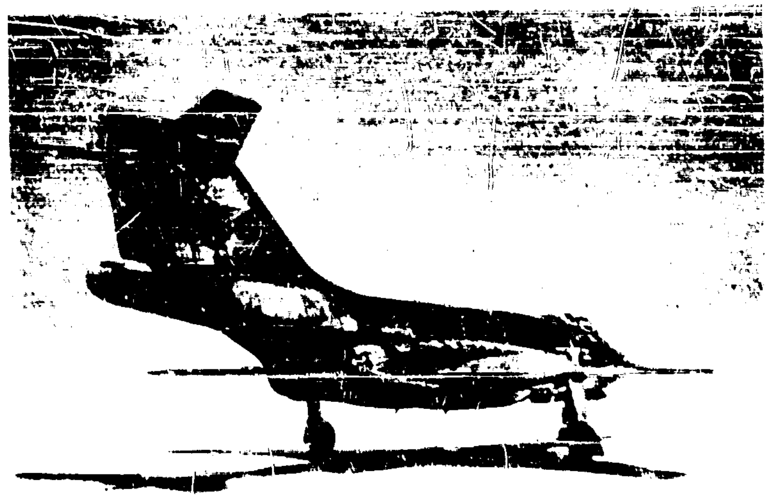
rear



right side

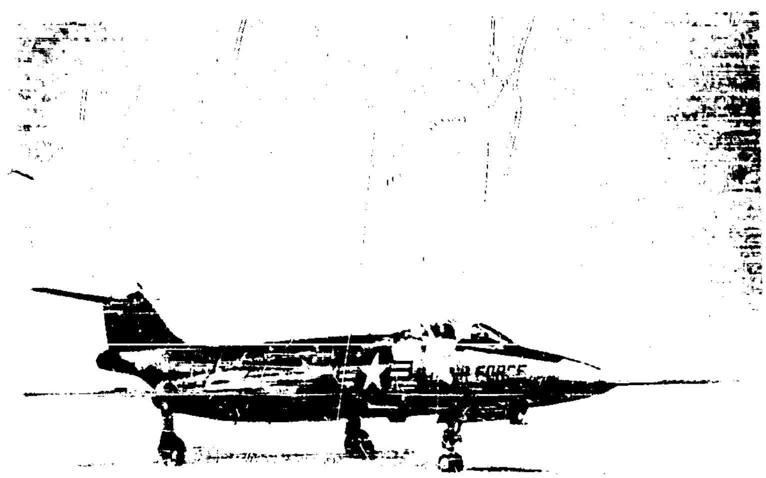


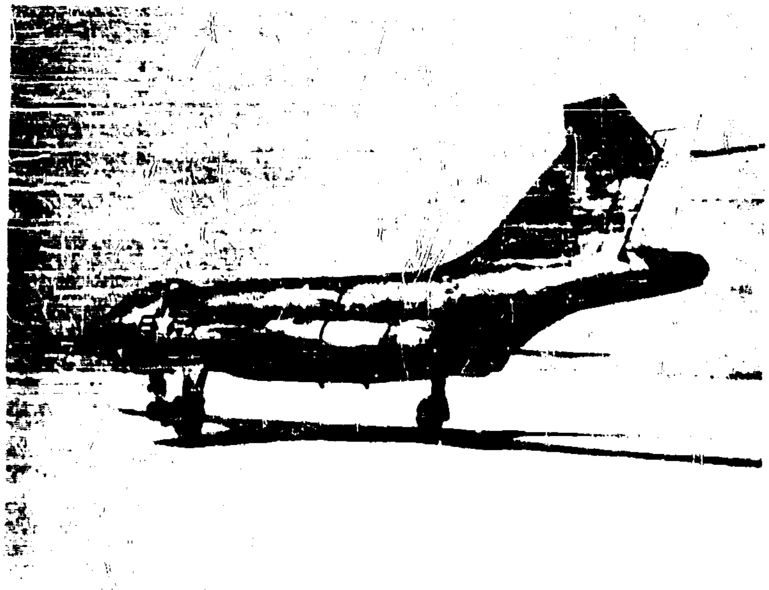
left side



right rear

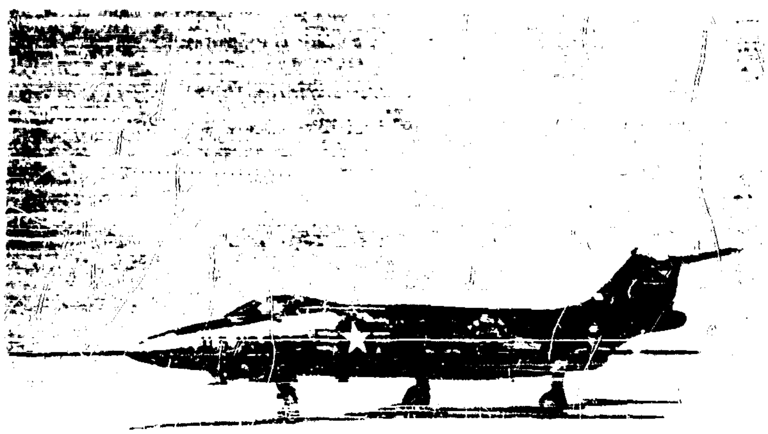
right front





left rear

left front



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